OFFICE OF ADVANCED RESEARCH AND TECHNOLOGY
SPACE VEHICLES DIVISION

PROCEEDINGS OF CONFERENCE ON

ACTIVE TEMPERATURE CONTROL

NASA

APRIL 9, 1964

NASA HEADQUARTERS

WASHINGTON, D.C.

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Table of Contents

List of Attendees

Conference Agenda

Conference Notes - Conrad P. Mook

"Active Thermal Control of the Micrometeoroid Capsule Electronics"

Tommy C. Bannister, MSFC

"Capillary Screen Pump in a Forced Convection System"

Thomas O. Thostesen, JPL

"The Use of Radiation Shields for Thermal Control on the Lunar Surface"

John C. Arvesen, Ames Research Center

"Discussion of the Oklahoma State University Active Temperature Control System"

Robert E. Kidwell, Jr., GSFC

"The Research and Development Program on Controllable a/e Ratio Surfaces for Temperature Regulation of Spacecraft"

William J. O'Sullivan, Jr., Langley Research Center

"Phase Change in Polymeric Systems for Active Temperature Control"

Warren C. Kelliher, George F. Pezdirtz, and Philip R. Young, LRC

"Research on Active Temperature Control at Goddard Space Flight Center"

Stanley Ollendorf, GSFC

"Temperature Control Louvers for the Mariner Venus and Mariner Mars Spacecrafts"

M. Gram, JPL

[redacted]

ATTENDEES FOR ACTIVE TEMPERATURE CONTROL CONFERENCE

April 9, 1964

NAME	CENTER
Warren C. Kelliher	Langley Research Center
Doyle P. Swofford	Langley Research Center
George Pezdirtz	Langley Research Center
William J. O'Sullivan, Jr.	Langley Research Center
Robert W. Johnson	Langley Research Center
Wayne S. Slemp	Langley Research Center
Robert Kidwell	Goddard Space Flight Center
Norman Ackerman	Goddard Space Flight Center
Archie Cunningham	Goddard Space Flight Center
Joe Skladany	Goddard Space Flight Center
Nelson Hyman	Goddard Space Flight Center
Michael Coyle	Goddard Space Flight Center
Stan Ollendorf	Goddard Space Flight Center
Michael E. Kahn	Jet Propulsion Laboratory
John W. Lucas	Jet Propulsion Laboratory
Thomas O. Thostesen	Jet Propulsion Laboratory
Elmer Christensen	Jet Propulsion Laboratory
John W. Orsag	Manned Spacecraft Center
Tommy C. Bannister	Marshall Space Flight Center
Bill J. Duncan	Marshall Space Flight Center
Gene Comer	Marshall Space Flight Center
Donald L. Anderson	Ames Research Center
John C. Arvesen	Ames Research Center
Herman Mark	Lewis Research Center
Arthur Reetz	NASA Headquarters
Conrad P. Mook	NASA Headquarters
Warren Keller	NASA Headquarters
Milton Schach	Goddard Space Flight Center

Space Vehicles Division
Office of Advanced Research and Technology
NASA Headquarters
Washington, D. C.

ACTIVE TEMPERATURE CONTROL

Agenda for Review and Planning Conference - April 9, 1964 to be held in OART Conference Room 6032, Federal Office Building 10B.

Thursday, April 9, 1964

9:00	Introduction and Welcome
9:15	Thermal Control of the MMC (Micrometeoroid Capsule) Electronics - Tommy C. Bannister - Marshall Spase Flight Center
9:45	Screen Capillary Pump - T. O. Thostesen - Jet Propulsion Laboratory
10:15	BREAK
10:30	Discussion and Demonstration of the Hughes Thermal Switch - Elmer Christensen - Jet Propulsion Laboratory
11:00	The Use of Radiation Shields for Thermal Control of Vehicles on the Lunar Surface - John Arvesen - Ames Research Center
11:30	Discussion of the Oklahoma State University Active Temperature Control System - Robert Kidwell - Goddard Space Flight Center
12:15	LUNCH
1:15	Spacecraft Temperature Control Research at Langley - William O'Sullivan, Jr. - Langley Research Center
1:45	Phase Change in Polymeric Systems for Active Thermal Control - Warren C. Kelliher, George F. Pezdirtz, and Philip R. Young - Langley Research Center
2:15	Research on Active Temperature Control at Goddard - Stanley Ollendorf - Goddard Space Flight Center
3:00	BREAK
3:15	Round Table Discussion of Active Temperature Control Systems for Manned and Unmanned Spacecraft Moderator - Conrad P. Mook

Conference Notes

The NASA conference on Active Temperature Control was one of a series held during the first half of 1964 in connection with the Thermal Radiation and Temperature Control Program.

The papers published herein comprise those presented on April 9, 1964 in accordance with the conference agenda. Mr. Cristensen's paper, though not published, was presented in a form which permitted notes to be taken by the undersigned.

Mr. Cristensen displayed a prototype version of the Hughes Thermal Switch which will be used on the Surveyor Lunar Lander. The switch when closed has a conductance of .5 BTU/hr. °F, which decreases to .0015 BTU/hr. °F when open. He explained that cold welding was avoided by spraying a .001" teflon layer on one contact. The switch weighs approximately one-half pound.

One additional paper, in addition to those shown on the agenda, was presented by Dr. John Lucas of JPL. This paper, a discussion of JPL louver systems, is summarized herein by M. Gram of JPL.

At the close of the conference a round table discussion was held, in which the matter of engineering development of the systems described by Mr. O'Sullivan was discussed and the matter was left to Langley Research Center to work out in cooperation with Headquarters.

It was pointed out that the Langley work was not duplicating that of Dr. John Schutt of GSFC in that the former considered changes of the step-function type.

C. P. Mook
NASA Headquarters

ACTIVE THERMAL CONTROL OF THE MICROMETEOROID CAPSULE ELECTRONICS

Introduction

Thermal louvers, similar to those on Mariner II, are to be used on the MMC electronics canister for thermal control. At the present time, it is felt that all louver applications, laboratory test results, and flight data are a necessary and complimentary part of louver development.

Figure 1 shows the spacecraft. Figure 2 shows the micrometeoroid panels being constructed. Notice the size of the man in the figure relative to the size of the spacecraft.

Thermal Design

The design philosophy is to locate all the temperature sensitive electronics components in an electronics canister and control the canister temperature with thermal louvers. The canister is shown in Figure 3, a thermal schematic of the spacecraft. The louvers are positioned so as to eliminate incident solar radiation. The heat transfer through the louvers is strictly infrared radiant transfer to the vehicle (as a sink). The vehicle coating will be a stable white paint.

The interior of the canister is shown in Figure 4. Special mounting techniques are employed for the extra-sensitive batteries to maximize the contact area.

Figure 5 shows the canister mock-up being used for the thermal vacuum tests now in progress. Two bays of louvers are attached to the lower face of the canister. The other faces consist of superinsulation sandwiched between two stiff panels.

An idea of the number of times the louvers are expected to actuate in orbit can be obtained from Figure 6, which shows typical curves for time in sunlight for the spacecraft. A minimum of four major actuations are expected, and probably more, depending upon attitude variations. One of the louver bays is shown in Figure 7. Each louver bay weighs approximately two pounds, has a total blade area of 1.7 ft^2, is actuated by bimetallic spirals (with each blade being independently actuated) whose temperature is radiatively linked to the components, and has a highly polished aluminum blade surface. The actuators are thermally isolated from the structure and sink by a special superinsulated housing with an open face towards the components. The free end of the blade rides in a teflon bushing to prevent cold welding and reduce friction. Fabrication problems have been encountered in adhering to the rigid tolerances imposed. Hand selection of parts has become necessary.

Thermal vacuum tests are presently being conducted to check the thermal design, especially in the operation of the louvers. Preliminary results show the design to be verified.

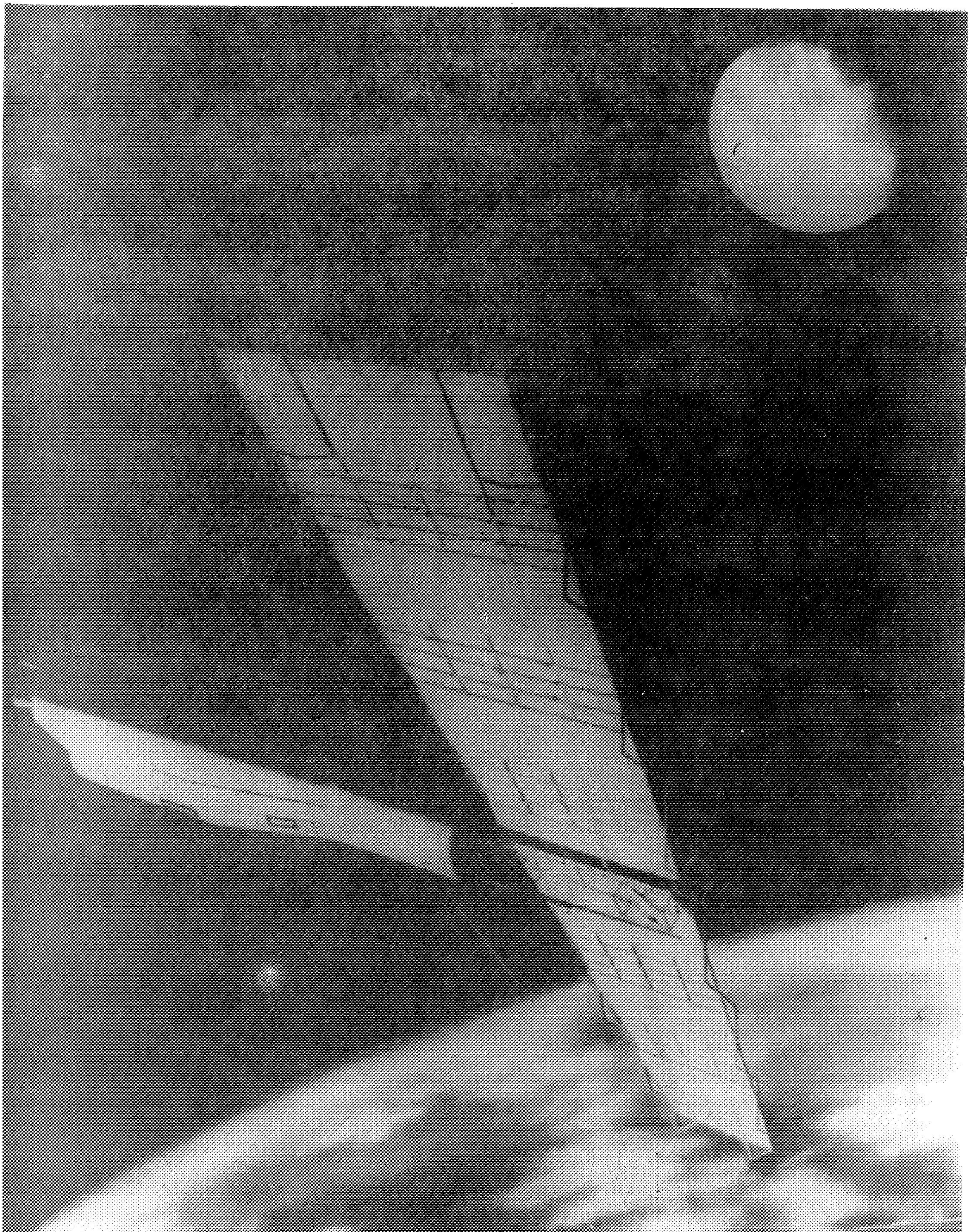


FIGURE 1

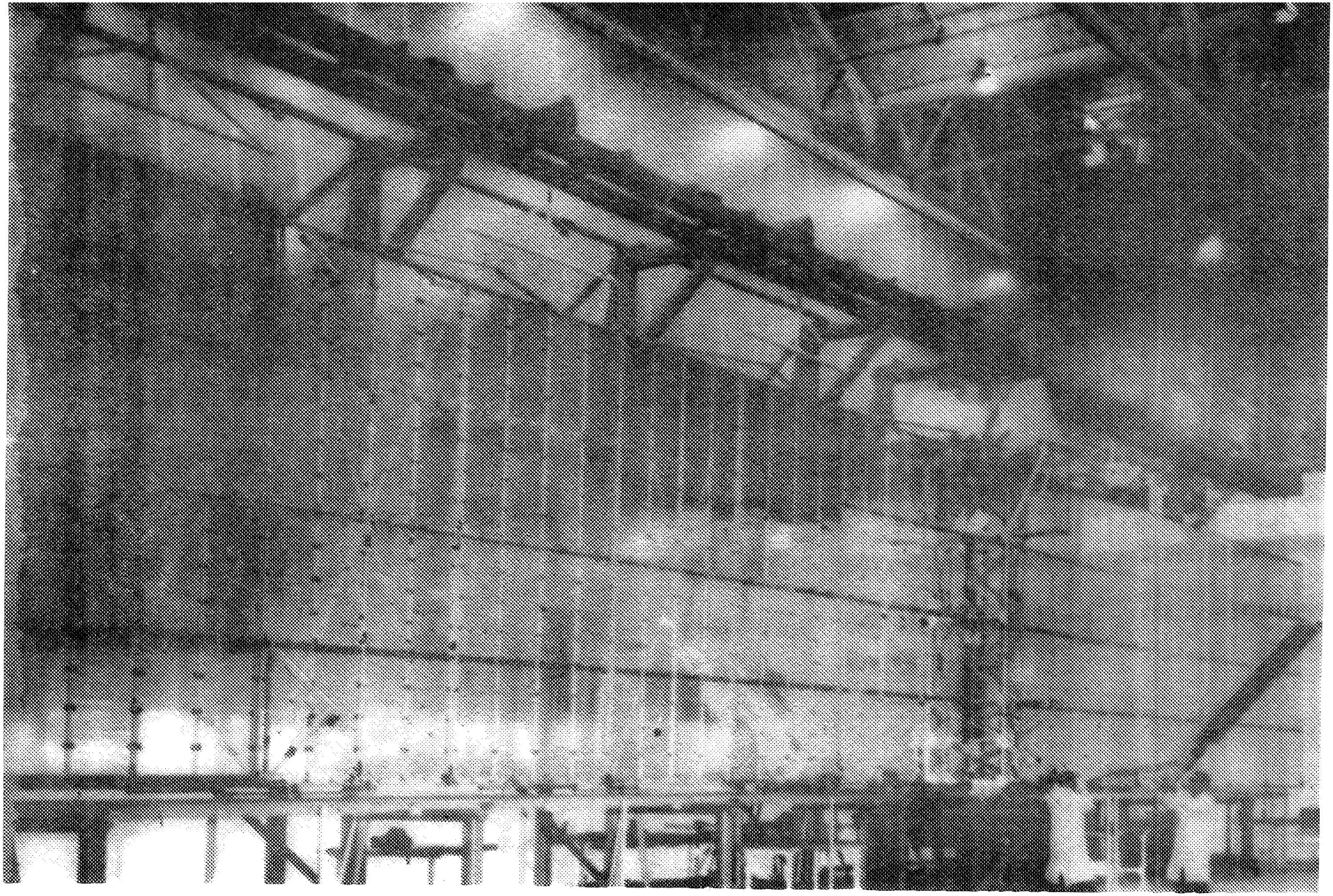


FIGURE 2

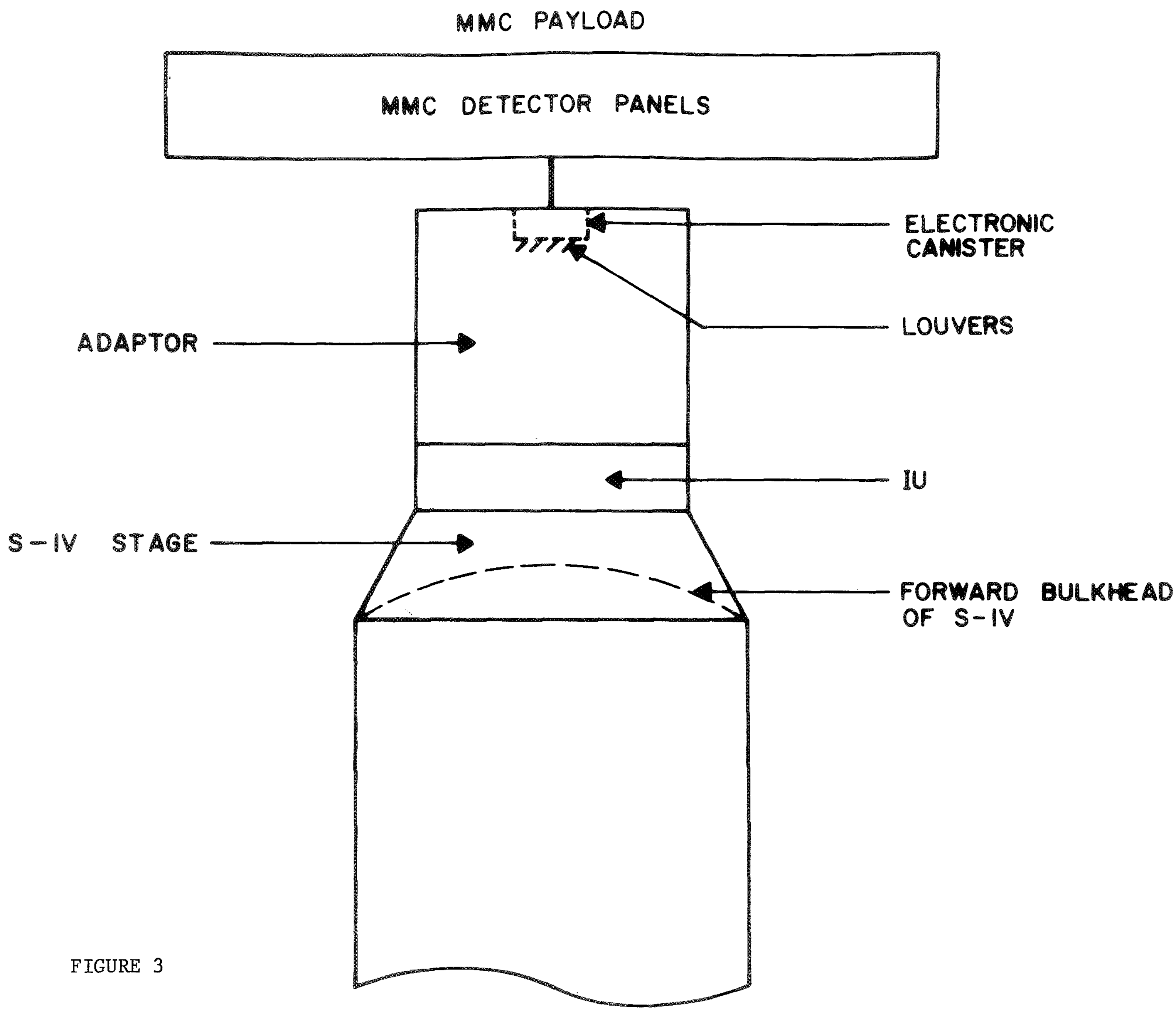

MMC PAYLOAD
MMC DETECTOR PANELS
ELECTRONIC CANISTER
LOUVERS
ADAPTOR
IU
S-IV STAGE
FORWARD BULKHEAD OF S-IV

FIGURE 3

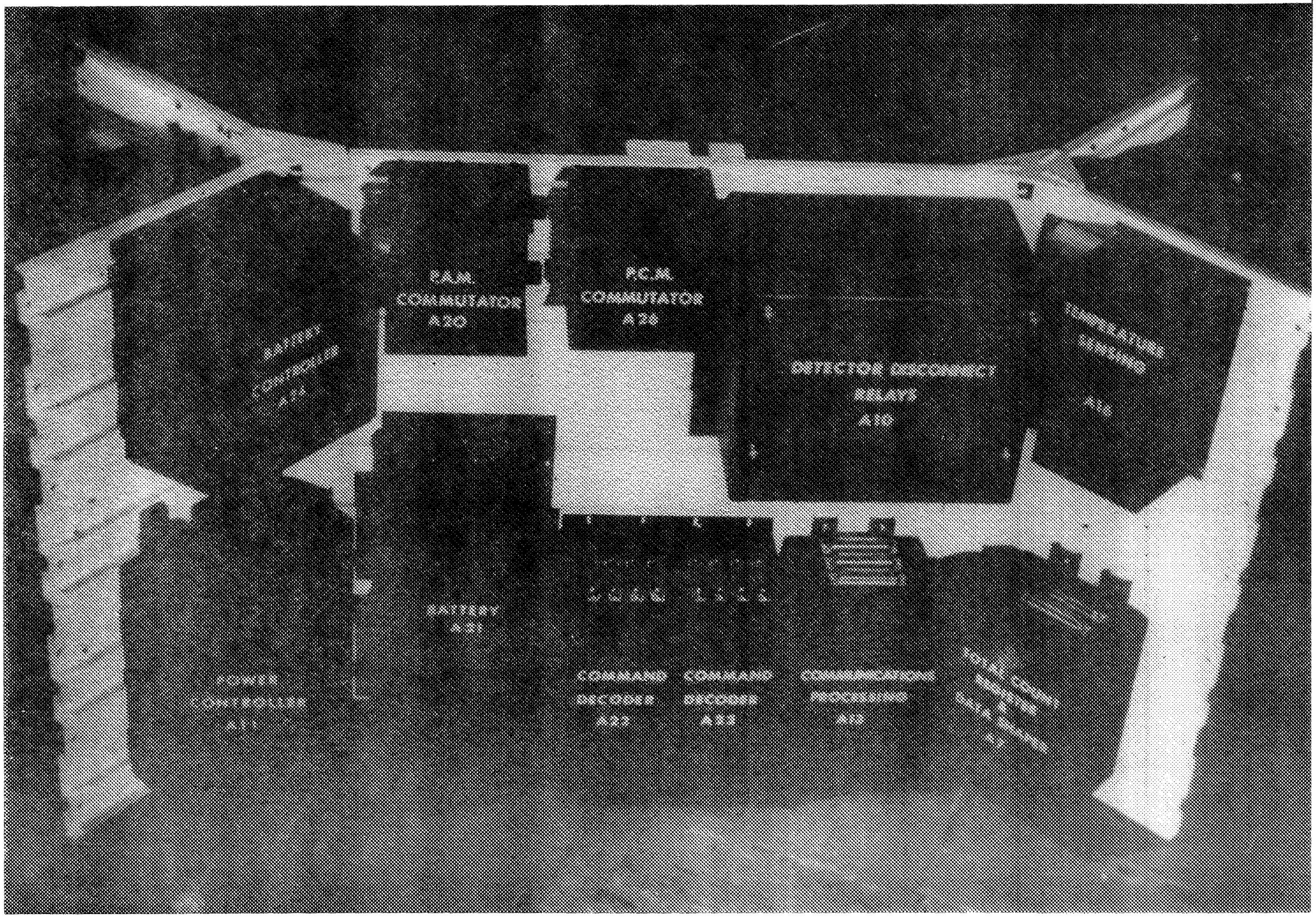

P.A.M.
COMMUTATOR
A20
P.C.M.
COMMUTATOR
A28
DETECTOR DISCONNECT
RELAYS
BATTERY
POWER
CONTROLLER
COMMAND
DECODER
COMMAND
DECODER

FIGURE 4

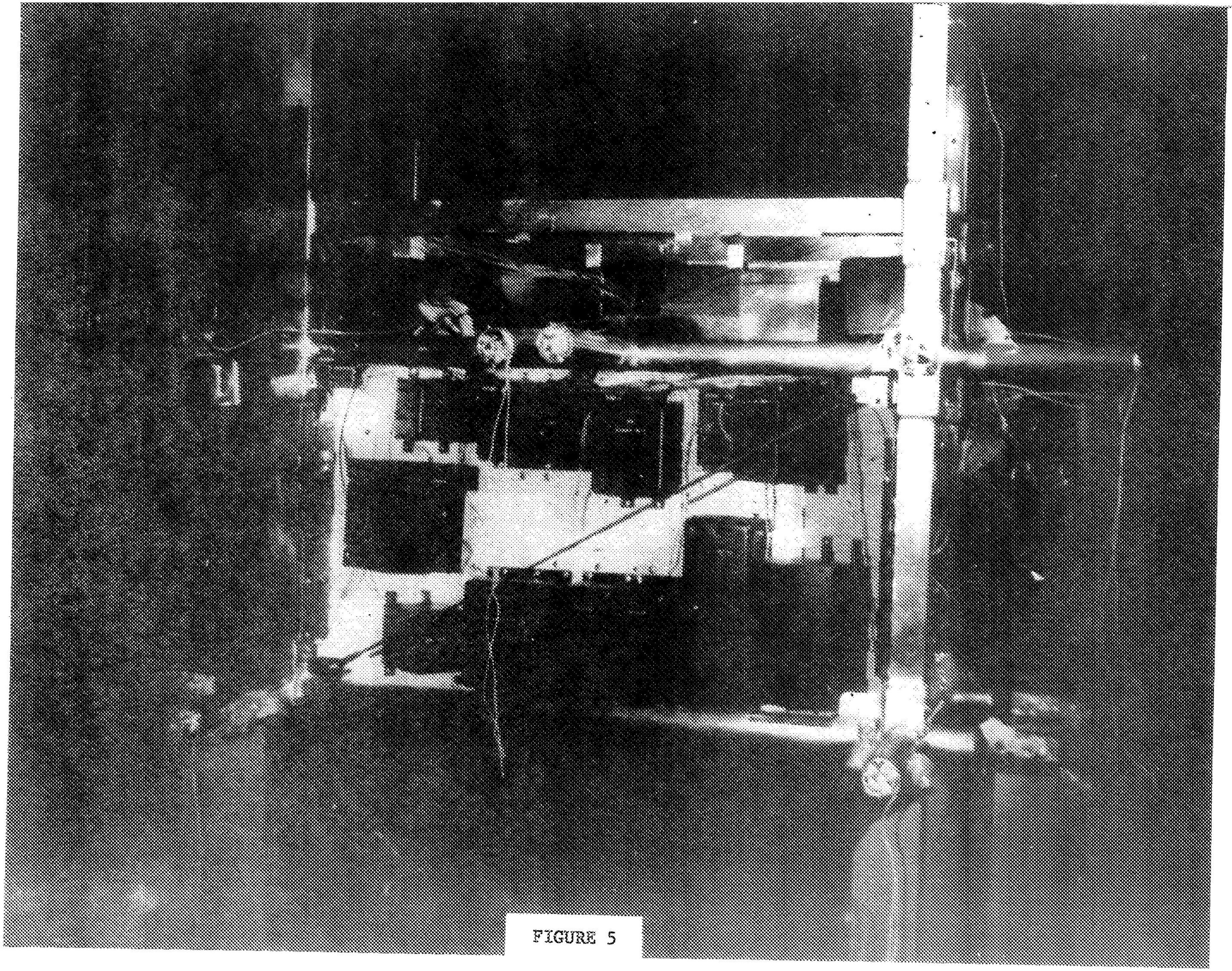
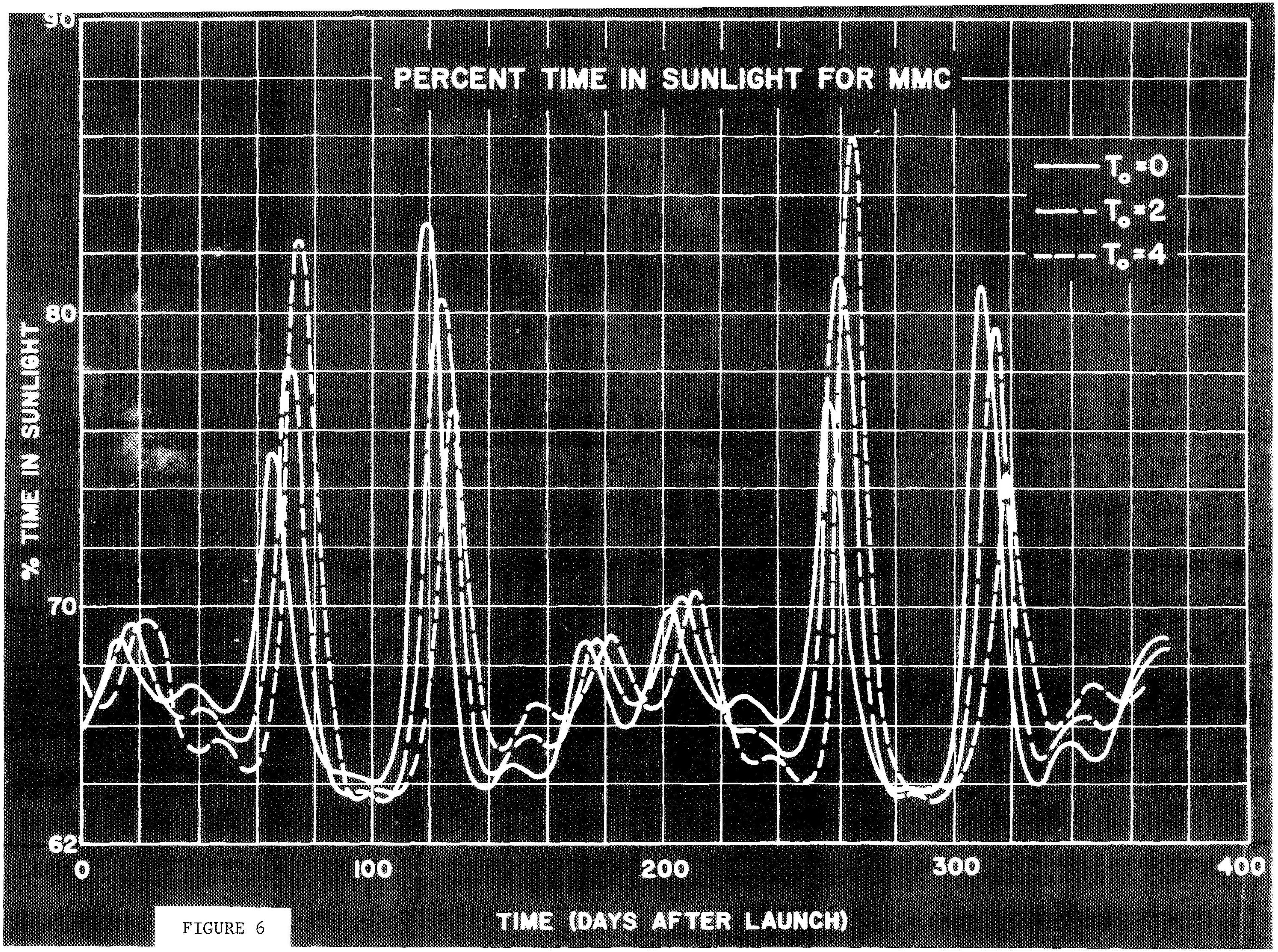


FIGURE 5



PERCENT TIME IN SUNLIGHT FOR MMC
T_o=0
T_o=2
T_o=4
% TIME IN SUNLIGHT
90
80
70
62
0
100
200
300
400
TIME (DAYS AFTER LAUNCH)

FIGURE 6

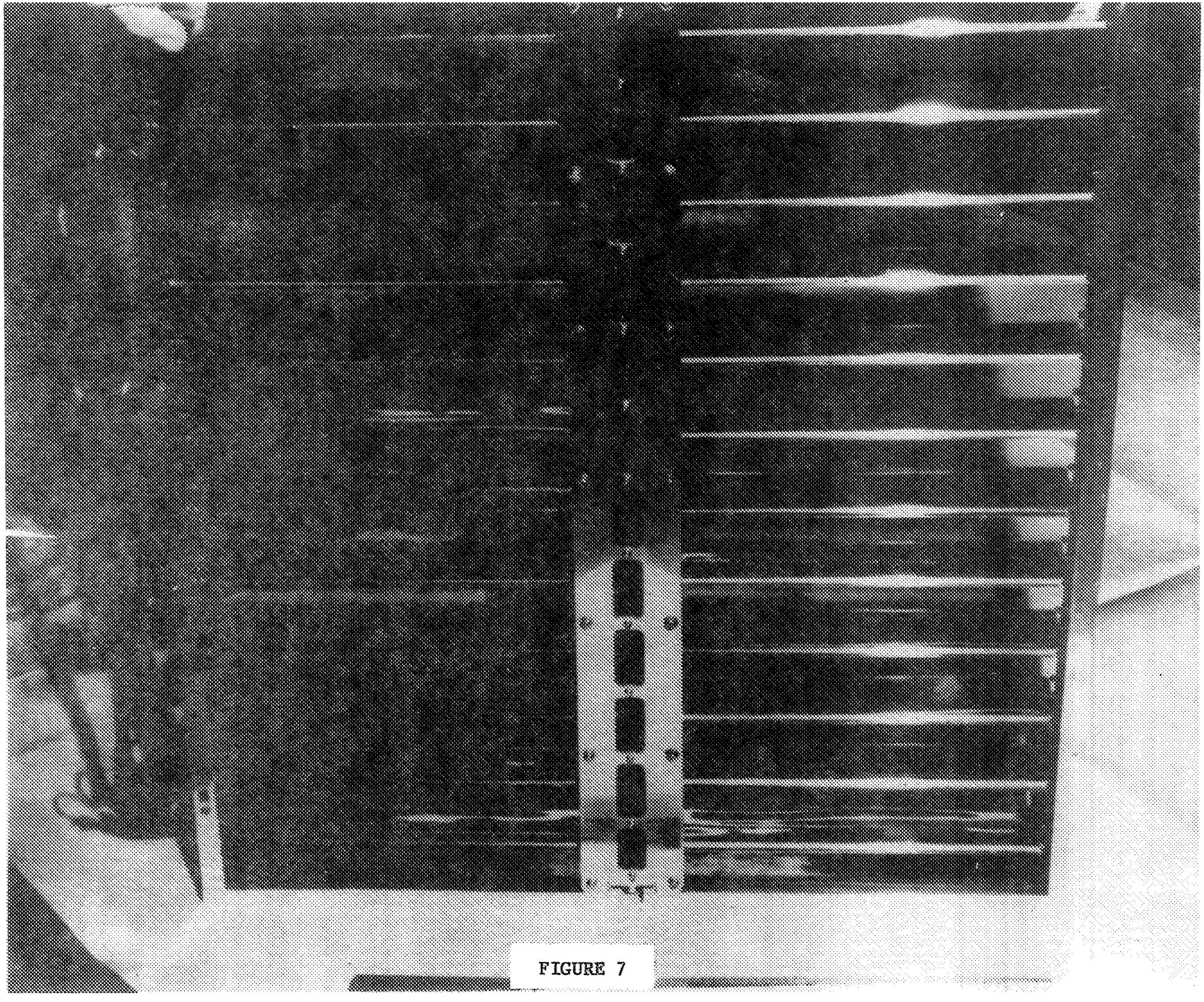


FIGURE 7

CAPILLARY SCREEN PUMP IN A FORCED CONVECTION SYSTEM

Thomas O. Thostesen

Jet Propulsion Laboratory

Concentrated internal power dissipations in large, densely packed spacecraft may require means of transferring heat in excess of the radiation and structural conduction paths arising from the packaging design. Either additional conduction paths or convection loops would then be necessary.

The present investigation is directed toward meeting the convective loop needs within the present restrictions of weight and power. The purpose is to evolve a two-phase heat transfer device which does not require mechanical or electrical power and will operate reliably for a period of years while withstanding the vibrations and accelerations of the spacecraft environment. The thermal energy would be transferred by evaporation of a fluid at the heat source and condensation of the vapor at the heat sink. The fluid would be transported by differences in vapor pressure and surface tension forces.

Figure 1 shows such a system employing a wick. The liquid is evaporated at the warm end of the wick, the vapor transfers to the cold end and condenses, and capillary forces replenish the liquid supply in the evaporator. Wicks are not necessarily the optimal system. Viscous losses, proportional to $\frac{v}{r^2}$, are quite large in miniscule passages. Since the capillary force is inversely proportional to r, the velocity in the tube is proportional to the hydraulic radius in case the system is not heat transfer limited. In this simplified case the system should have passages as large as are consistent with the vibrational environment.

One method of reducing the viscous forces is to eliminate the wicking in the piping which joins the condenser and the evaporator. The next area to investigate is the wicking in the evaporator and condenser. The phase change occurs just at the surface of the wicking (or at the meniscii - Fig. 2). An analysis of the heat transfer shows that most of the heat of vaporization is in the immediate vicinity of the meniscus. This would suggest that the ideal system would provide just the meniscii and eliminate the constrictions present in wicks.

The proposed system is shown in Figure 3. Here the fluid is retained and moved by the surface forces generated in the meniscii formed by a screen mesh. Evaporation and condensation occur at the surface of the liquid, with the heat flowing between the surface and the source or sink. Figure 4 shows in more detail the mechanism of the screen meniscii. The pressure differential across a meniscus is inversely proportional to the radius of curvature. In a capillary, this differential is constant as the meniscus changes position (4.a.) In the screen system, the radius of curvature changes as the meniscus moves (4.b. and 4.c.). Therefore, the pressure differential is a function of the amount of liquid in the system at that point. As the meniscus recedes, more "suck" is applied to the liquid. 4.d. illustrates that woven screening is not as simple a situation as parallel rods or a toroid.

Preliminary tests with water as the fluid were run at one atmosphere to determine approximate capabilities. The screen unit used a 4" diameter 150 mesh (wires per inch) stainless steel screen separated from the source or sink plate by 1/32". It was found that this unit would:

a) Support a head of 4" of water in suction.

b) Evaporate water at the thermal rate of 20 watts at 156°F against zero head.

c) Condense water at the thermal rate of 75 watts with the condensate being drawn off by a pressure differential of 2" of water.

d) Withstand 40g rms vibrations from 100 to 2000 cps.

A test with the vapor space evacuated of noncondensable gas showed that this causes the vapor flow to be maximized.

Based upon the preliminary data, a design goal for a closed cycle, evacuated unit was established as 20 watts for 6 hours with a temperature differential between source and sink of 50°F over a distance of 1 foot with evaporator and condenser diameters of 4". These goals are being approached by investigating the evaporator and the condenser separately. When the two components perform satisfactorily, they will be combined into a closed system.

Tests on the evaporator to date have not been too encouraging, but hopes are high. Early experiments were plagued with bubbles forming in the supply line from the reservoir to the evaporator, choking off the flow. This was eliminated by replacing Tygon lines with glass tubing and surgical rubber tubing. The maximum evaporation rate in a vacuum has been 3 watts, with the limitation imposed by catastrophic evaporation behind the screen in the liquid space. It is expected that this is a dissolved gas problem and better deaeration techniques will be the answer.

The condenser side of the system has not yet been tested in a

vacuum environment but should not be a problem, since there is much less probability for bubble formation.

As noted, bubble formation in the liquid phase is a serious problem. A bubble which has sufficient volume to contact the inner circumference of a tube blocks that tube against passage of liquid by capillary forces. A bubble behind a screen blocks off part of the screen. Once the bubble forms, it easily grows, fed by vapor from evaporation. Care in deaeration and material selection have and will reduce this problem greatly. In addition, bubbles behind the screen have disappeared, probably due to rupturing of a meniscus in the screen so that local capillary forces could expel the vapor through the opening.

The launch of a spacecraft might be violent enough to cause the liquid and vapor phases to become intermixed, requiring that the system be charged following launch. However, preliminary tests have shown the system to be quite resistant so that the unit might be charged before launch.

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3. T. O. Thostesen, "Screen Capillary Pump in a Forced Convection System," JPL SPS No. 37-23, Vol. IV, p. 58, October 31, 1963.

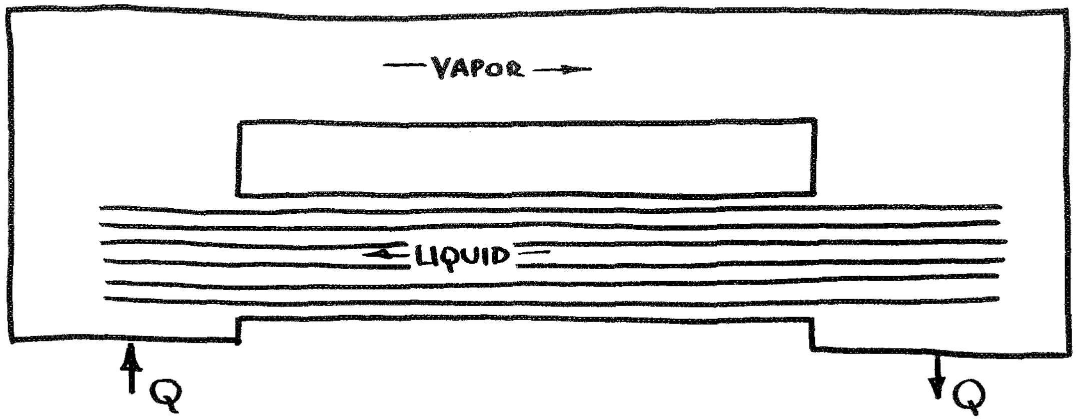

FIGURE 1

CAPILLARY WICK PUMP

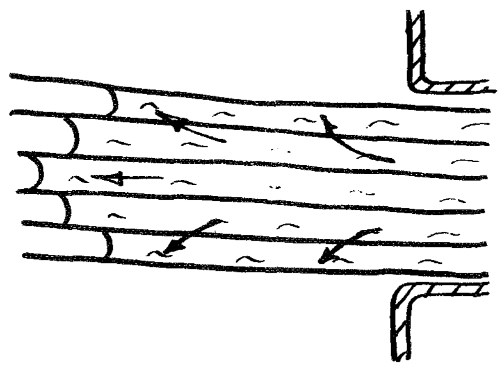

FIGURE 2

FLOW IN THE END OF A WICK

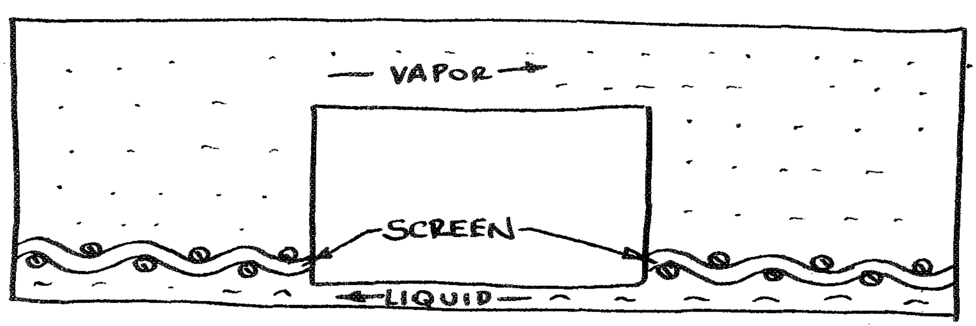

FIGURE 3

CAPILLARY SCREEN PUMP

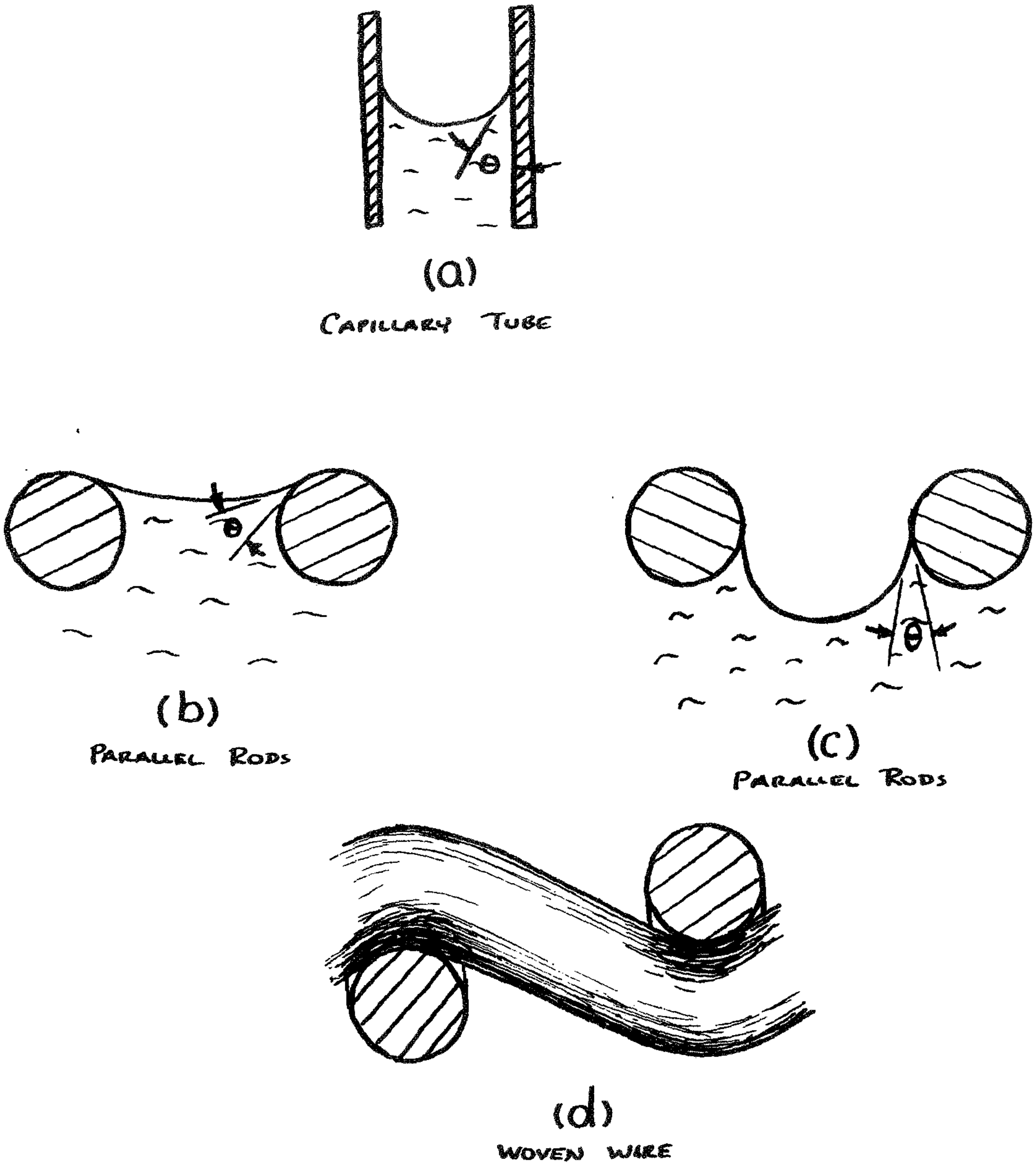

Figure 4

Meniscus Shapes

N66 32949

THE USE OF RADIATION SHIELDS FOR THERMAL CONTROL OF VEHICLES ON THE LUNAR SURFACE

John C. Arvesen
Ames Research Center
Moffett Field, California

INTRODUCTION

For the past several years Ames Research Center has been involved in determining methods of thermal protection for vehicles that have to travel very close to the sun, such as a Solar Probe. One thermal control method that has proven to be very effective is the use of solar radiation shields. It is known that a vehicle traveling from the earth to within a tenth of an astronomical unit from the sun will be exposed to an intensity level of over 100 solar constants. It was shown in reference 1 that if this vehicle is shielded from direct solar radiation, its temperature change can be held to less than 10° F. An unshielded vehicle, by comparison, would increase in temperature hundreds to thousands of degrees, depending upon the solar absorptance and emittance properties of its surface. Since shields were so effective in isolating this particular vehicle from solar radiation, a study was undertaken to extend a similar concept to the protection of vehicles on the surface of the moon. The complete results of this study are reported in reference 2 and the main points will be presented in this paper.

LUNAR ENVIRONMENT

Much is known about the thermal environment of the moon as a result of astronomical measurements. The temperature variation and the reflectance and emittance properties are known fairly accurately. It is also known that the thermal conductivity of the surface is lower than any natural material found on earth. This last fact, along with the complete absence of any atmosphere, makes thermal radiation the primary mode of heat transfer for vehicles on the lunar surface.

Probably the most widely accepted lunar surface temperature measurements have been made by William Sinton and his associates at Lowell Observatory. These measurements are shown in figure 1.

This figure shows the temperature variation at the equator during one complete lunar day. Even though the solid curve is an approximation that assumes zero thermal inertia of the surface material, it agrees well with the measured temperature points. The most significant fact to be noted is the extremely large temperature variation during the day. At the subsolar point the surface temperature is about 240° F. As the day goes on, the temperature drops until it reaches a value of -240°F after the sun sets.

UNSHIELDED VEHICLE

During the daytime a vehicle will be receiving a large amount of radiation from the hot lunar surface. This radiation level is particularly high if the vehicle is at or near the subsolar point. Other major sources of heat that will contribute to the temperature of a vehicle on the lunar surface are direct solar radiation, solar radiation reflected from the lunar surface, and internally generated heat.

Direct solar radiation is one of the largest heat inputs to the vehicle. For vehicles with a high ratio of solar absorptance to emittance, the level of direct solar radiation will probably be the main factor in determining the vehicle's temperature. Solar radiation that is reflected from the lunar surface onto the vehicle will usually be of secondary importance, since the reflectance of the lunar surface material is only about 7 percent. For vehicles with a low α_s/ϵ coating, such as white paint, the main heat inputs will be radiation emitted from the lunar surface and internal heat. In order to evaluate the effects of these heat inputs on the temperature of a lunar vehicle, a particular configuration was chosen for study. This vehicle is shown in figure 2.

The design of the vehicle is very simple but the results obtained may be applied to more complex ones. The vehicle consists of a 60° conical section on top of a cylindrical body. It is assumed to be resting on a non-conducting support so that the only mode of heat transfer between it and the lunar surface is by radiation. There is also assumed to be no heat conduction between each section of the vehicle.

Subsolar Point

When the vehicle is at the subsolar point the conical section will receive direct radiation from the sun while the cylindrical

section will have no direct solar radiation incident. However, the cylindrical section will be exposed to radiation leaving the lunar surface to a greater degree than the conical section. The resulting temperature on each section for no internal heat is shown in figure 3. In this figure the surface temperatures are shown as a function of the vehicle's α_s/ϵ ratio, which varies from 0.1 to 10.0. Since the conical section has such a high level of solar radiation incident upon its surface, its resulting temperature is very dependent upon this α_s/ϵ ratio. For materials such as polished metals, temperatures may reach 300° F to 400° F.

The temperature of the cylindrical section, on the other hand, is relatively independent of the α_s/ϵ ratio since very little solar radiation is incident. It should be noted, however, that just the heat received from the lunar surface is enough to give an undesirably high temperature on this section, regardless of what its surface properties are. It should also be noted that unless a very stable white coating is used on the conical section its temperature will also be high.

Side Illumination

A unique situation is encountered if the sun is on the horizon, as would be the case if it were lunar dawn, dusk, or if the vehicle were at either pole. One side of the vehicle would receive the full intensity of the direct sun, while the other side would be in darkness.

If the heat conduction in the skin of the vehicle is assumed to be zero, the temperature distribution around the circumference of the cylindrical section will be as shown in figure 4. It can be seen that severe thermal gradients and large temperature differences may occur between the sunlit and shaded sides, especially at the higher values of α_s/ϵ. Heat conduction in the skin of the vehicle will help the situation but the effect is not really significant, as can be seen in figure 5. In this figure the vehicle is assumed to be 10 feet in diameter with an α_s/ϵ ratio of 1.0. The upper dashed curve represents a relatively high conductance skin of 1/8-inch thick aluminum. The other curves represent fractions of this value down to zero. The main effect is a temperature increase on the shaded side of the vehicle. However, the sunlit-side temperature remains relatively unchanged and there is still a large temperature difference from this side to the shaded side.

SHIELDED VEHICLE

Three different shield configurations were considered for use at the subsolar point and are shown in figure 6. The small shield (figure 6(a)) is the same diameter as the vehicle and does not shade the lunar surface. The next configuration (figure 6(b)) has a much larger shield which will shade and cool the surface of the moon within its shadow. The heat input from the lunar surface may be reduced by means of a shield that is symmetric around the base of the vehicle (figure 6(c)). Radiation emitted from the lunar surface will strike the shield and be reflected back. This shield is completely shaded by the large solar shield so that no direct sunlight will hit it. All these shields could be made fairly lightweight and might consist merely of a coated plastic film. The surface properties of these shields were found to be not particularly important so that if they were to undergo degradation there would be only a secondary effect upon the vehicle's temperature.

It is difficult to calculate the temperatures of a shielded vehicle because of the large number of surfaces involved. It must be considered that each surface could have different values of solar absorptance and emittance, as is the case for real materials. In order to have reasonable accuracy, infinite reflections between all surfaces must also be included. In reference 2 a general analysis was developed and was programmed for solution on an IBM 7090 computer. This computer program was then used to determine the effectiveness of the various shield configurations in reducing the temperatures of the two sections of the vehicle.

Subsolar Point

The temperatures on the conical section of the shielded vehicle are shown in figure 7(a). The unshielded case, as shown before, is very dependent upon the α_s/ϵ ratio and will reach temperatures that are much too high unless a very low ratio is used. For all shielded configurations the amount of solar radiation reaching this section has been almost entirely eliminated and, as a result, the temperature is almost independent of the α_s/ϵ ratio. Even at high values, the temperature is down to a reasonable level. This is important because it means that the surface of the vehicle could degrade badly from some environmental effect and the temperature would be unaffected.

The effect of shields upon the temperature of the cylindrical section is shown in figure 7(b). At the subsolar point the small

solar shield, which is the same diameter as the vehicle, has no effect on the cylindrical section. When the larger shield is added, the lunar surface around the vehicle will be shaded and cooled and the temperature of the cylindrical section will be correspondingly reduced. The main heat input is now from the hot lunar surface. By adding the lunar radiation shield around the base of the vehicle even this component can be effectively eliminated. Thus, by using shields, the temperature of this section <u>at the subsolar point</u> can be reduced to below -100° F. This level is probably too low for most applications but it should be apparent that the shields can be easily arranged to give almost any desired temperature between the unshielded and completely shielded limits. Internally generated heat will, of course, raise all temperatures.

Side Illumination

It was previously shown that a vehicle may have a large temperature difference from one side to the other when the sun is near the horizon. A solar shield is particularly useful here as can be seen in figure 8. In this figure, a single shield is placed between the vehicle and incident solar radiation. The temperature of the shielded side and the back side of the vehicle is shown as a function of the shield's separation distance. As the shield is moved farther away, the heat radiated to the vehicle becomes less and less until it approaches the level on the back side of the vehicle. Again, internal heat can be used to maintain the desired temperature level.

CONCLUDING REMARKS

This paper has attempted to show how effective shields can be in eliminating some of the problems that may be encountered when vehicles are on the surface of the moon. There is also no reason why a shield cannot be used by a man outside of the vehicle to reduce the solar and infrared heating of his spacesuit. In effect, solar shields are nothing more than the parasols that women have been using for years. The simplicity and versatility of radiation shields makes them an ideal safety measure, for both man and vehicle, in case of some unforeseen emergency. In conclusion, shields are an answer to most situations where a high degree of isolation from an external thermal environment is required.

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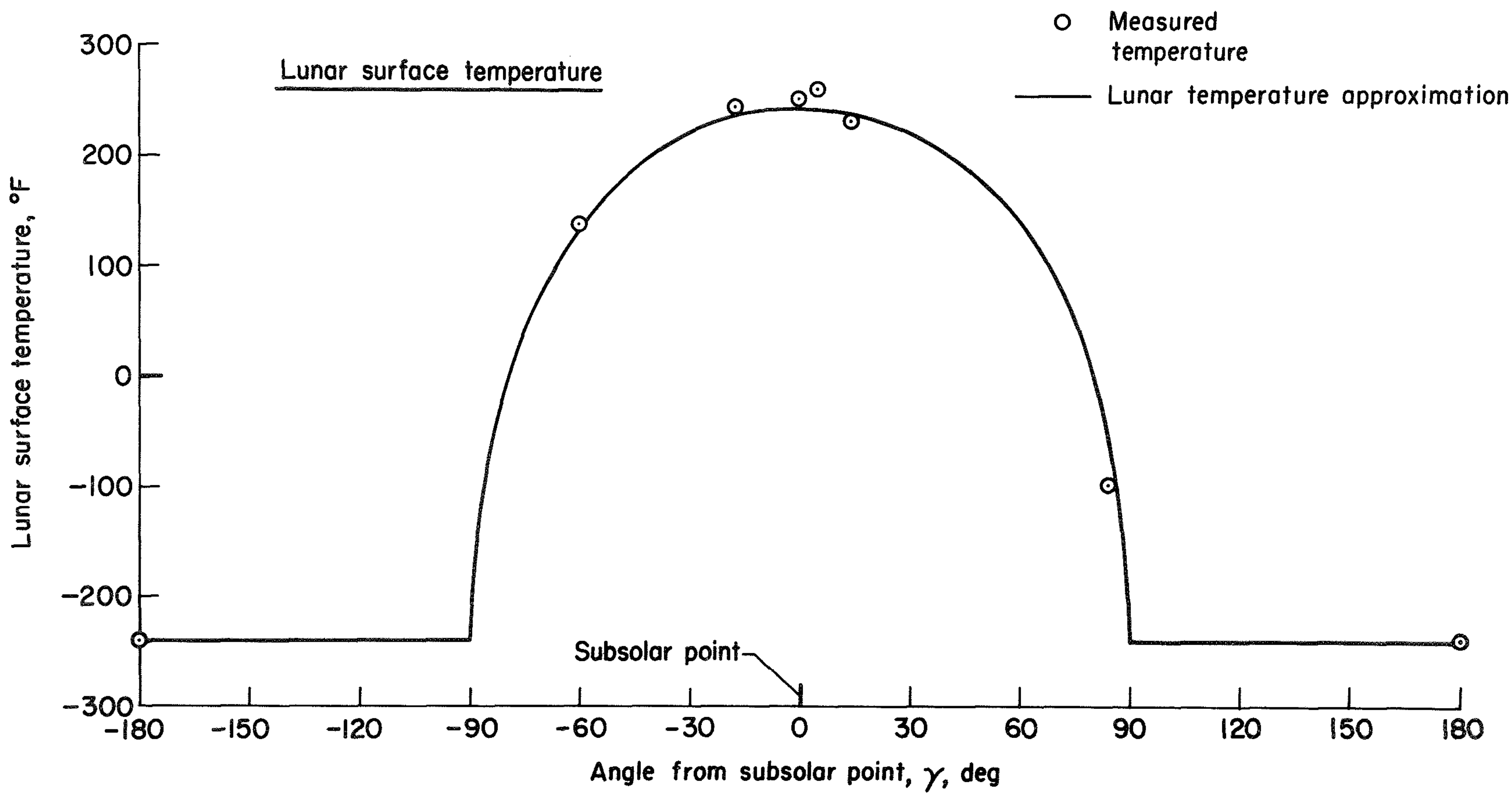

Figure 1. - Temperature variation of the lunar surface.

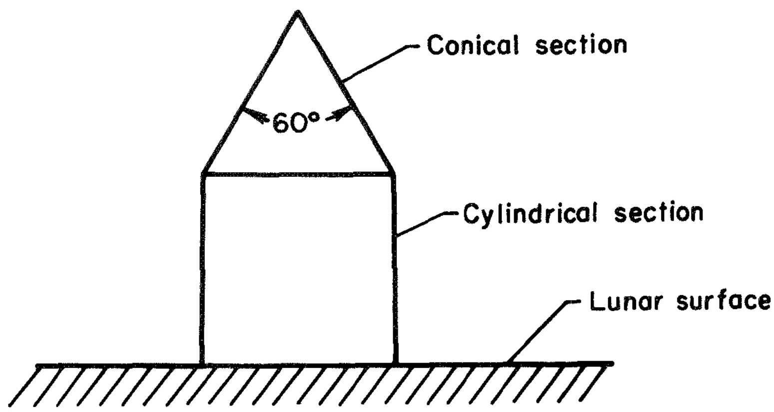

Figure 2. - Basic vehicle configuration.

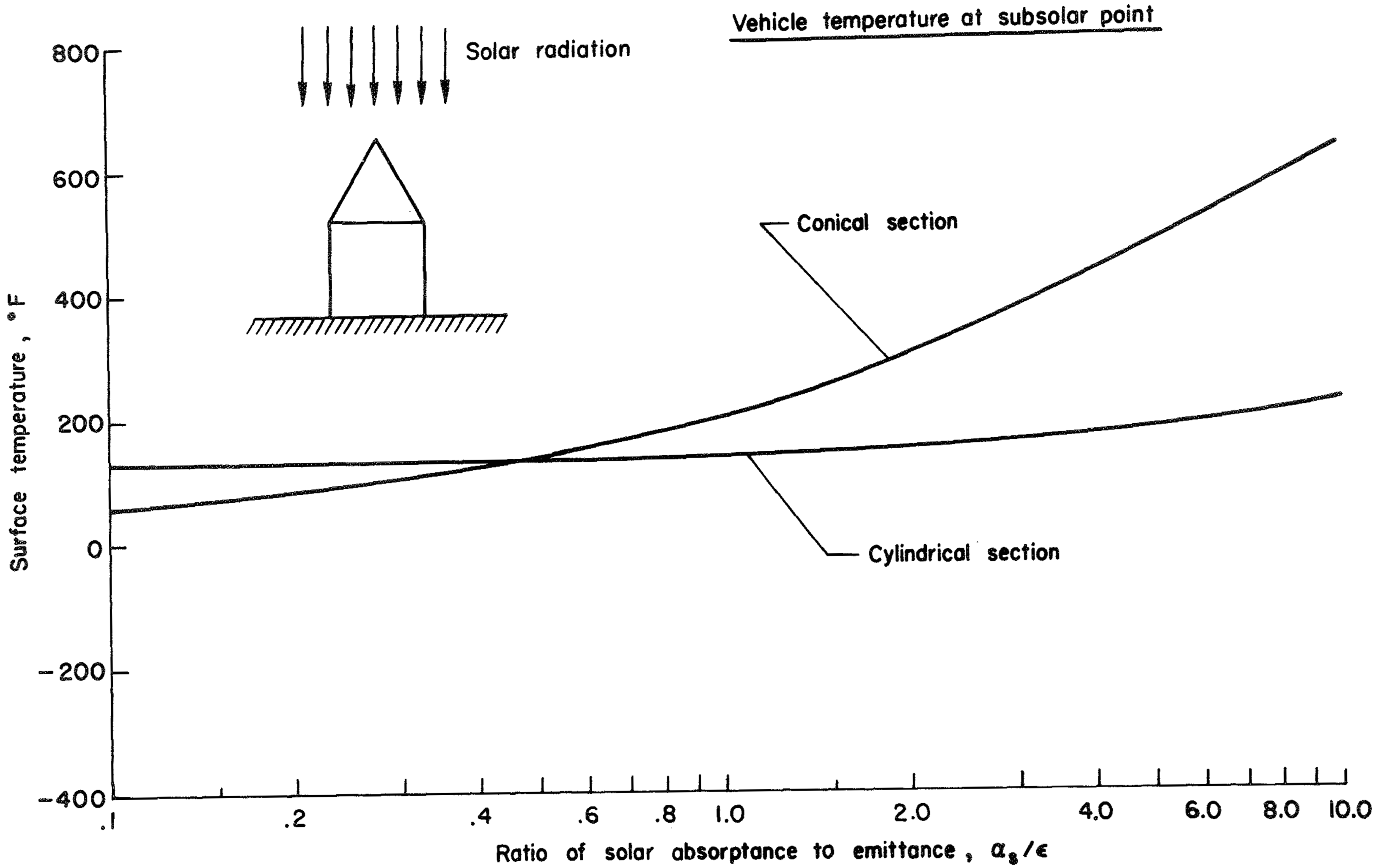

Figure 3. - Effect of the ratio of solar absorptance to emittance on the surface temperatures of the unshielded vehicle at the subsolar point; conductance and internal heat both zero.

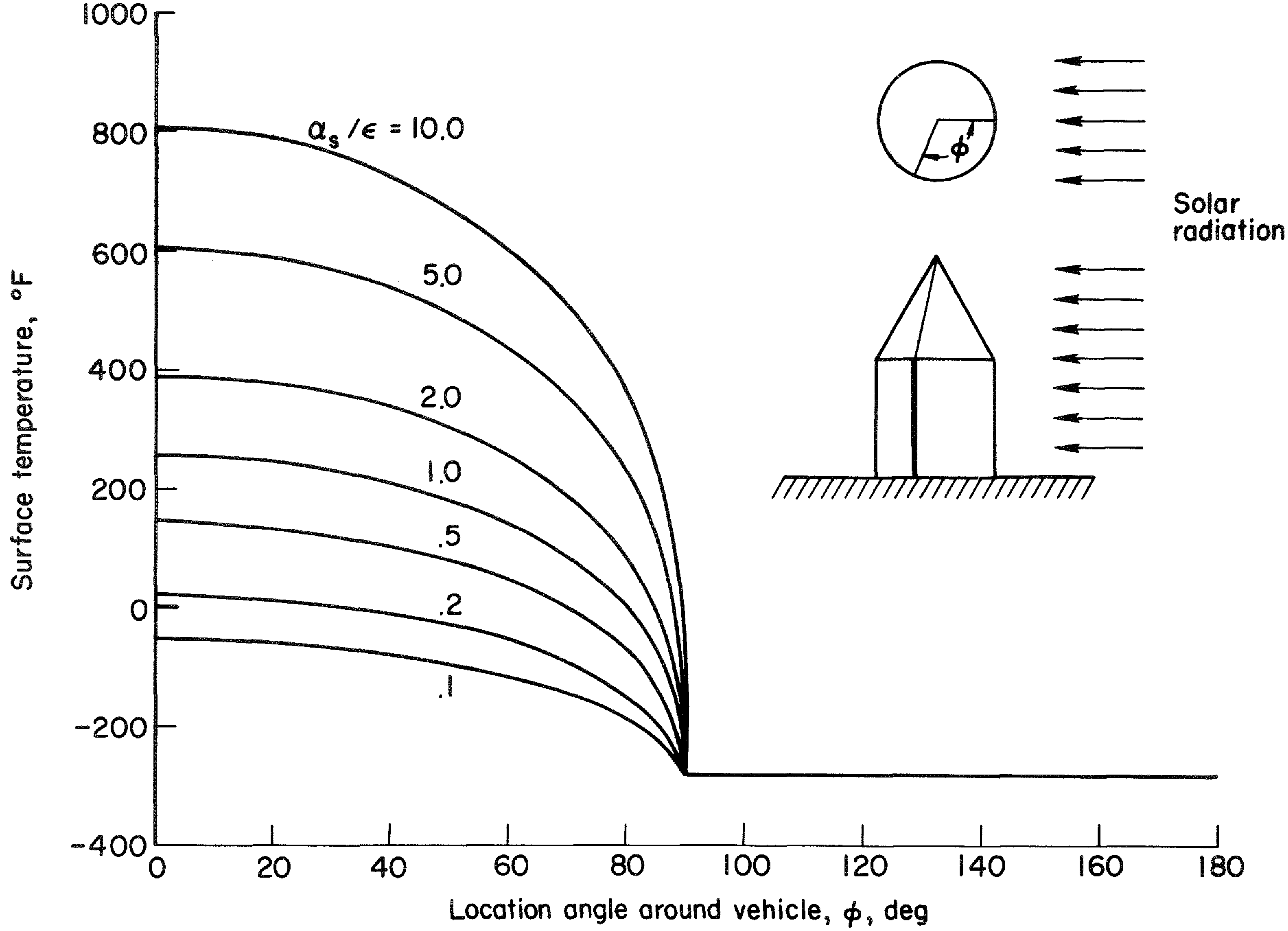

Figure 4. - Temperature variation around the cylindrical section of the unshielded vehicle when illuminated from the side; conductance and internal heat both zero.

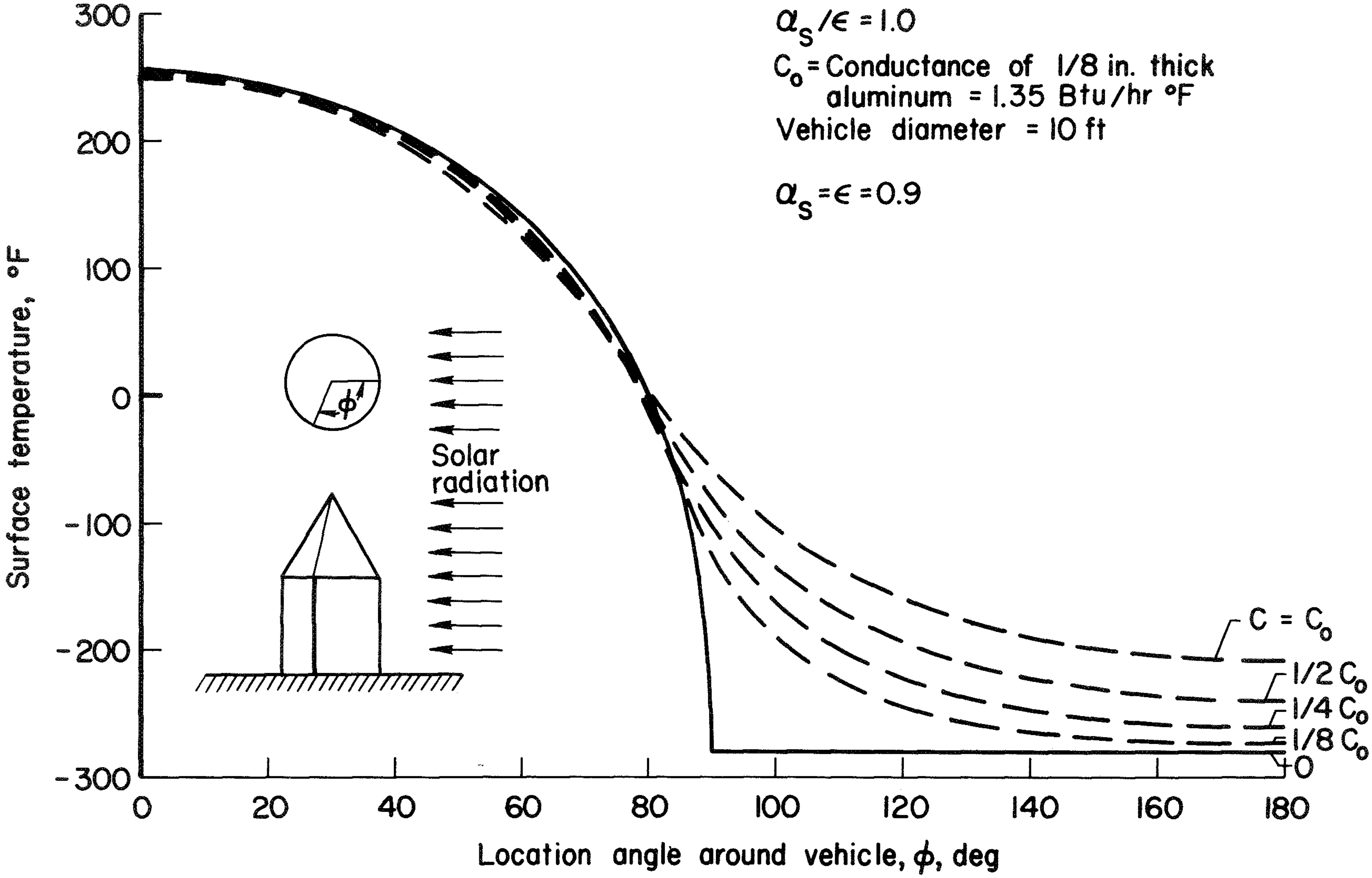

Figure 5. - Effect of thermal conductance upon the temperature variation around the cylindrical section of the vehicle when illuminated from the side; no internal heat.

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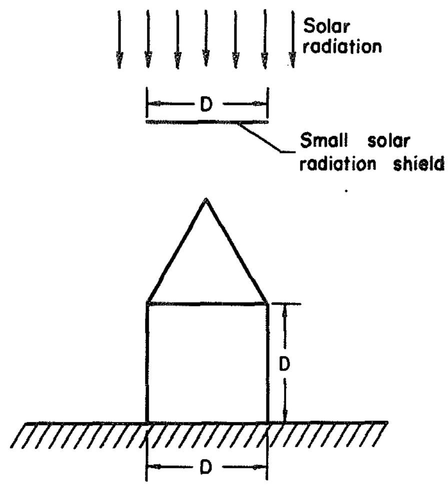
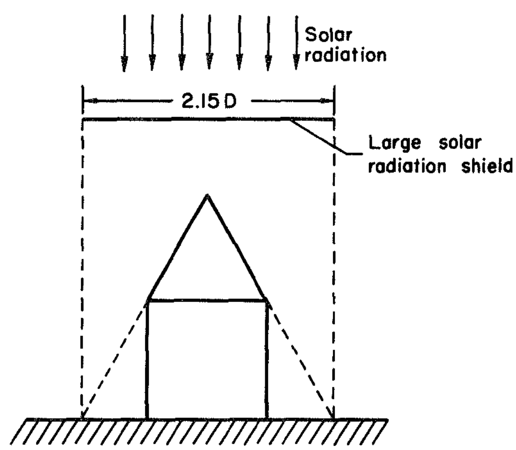
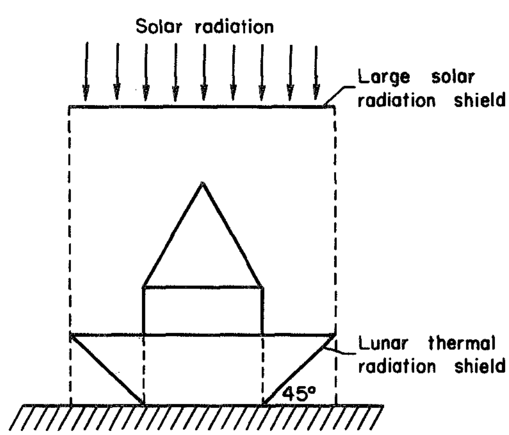


(a) Small solar shield

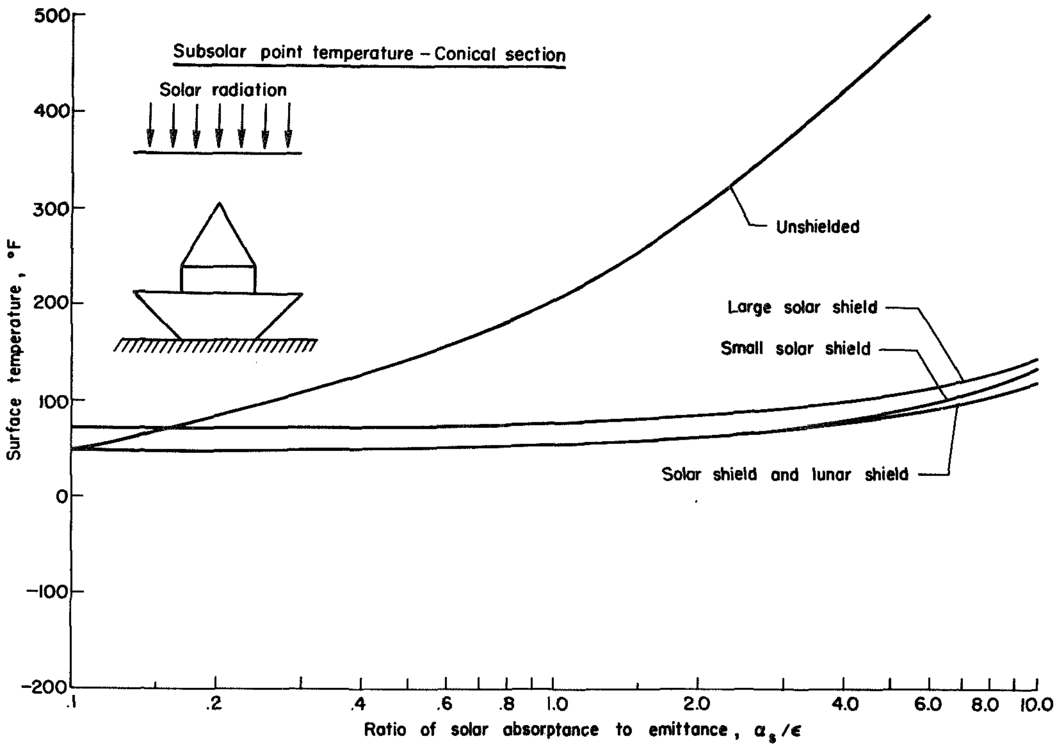


(b) Large solar shield

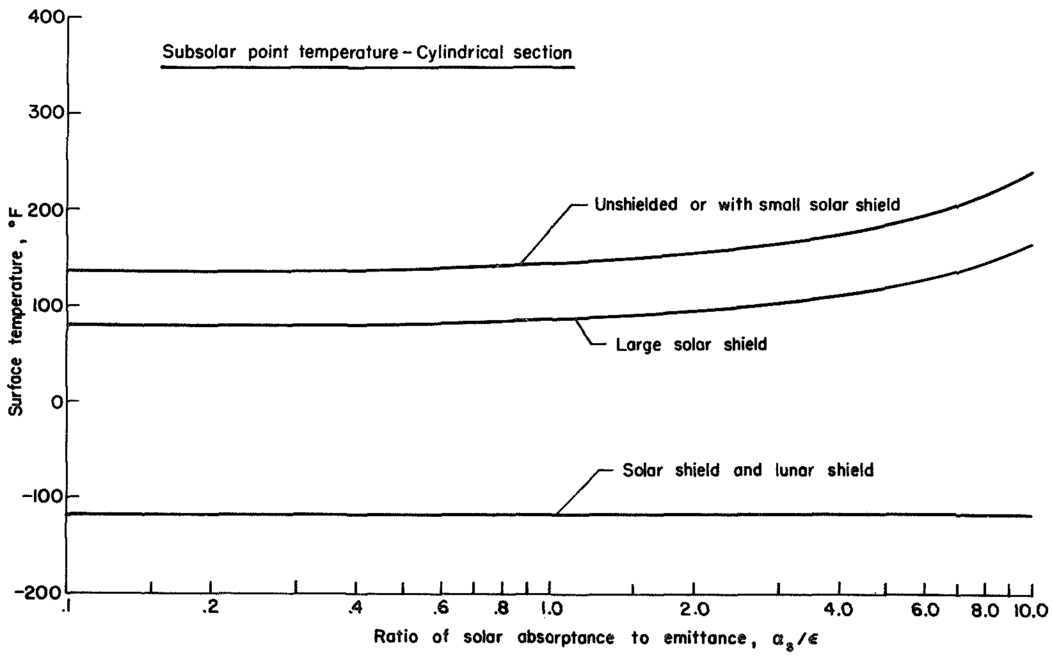


(c) Solar shield and lunar shield

Figure 6. - Shielded vehicle configuration



(a) Conical section



(b) Cylindrical section

Figure 7. - Effectiveness of solar and lunar radiation shields at the subsolar point as a function of the α_s/ϵ ratio on the vehicle; conductance and internal heat both zero.

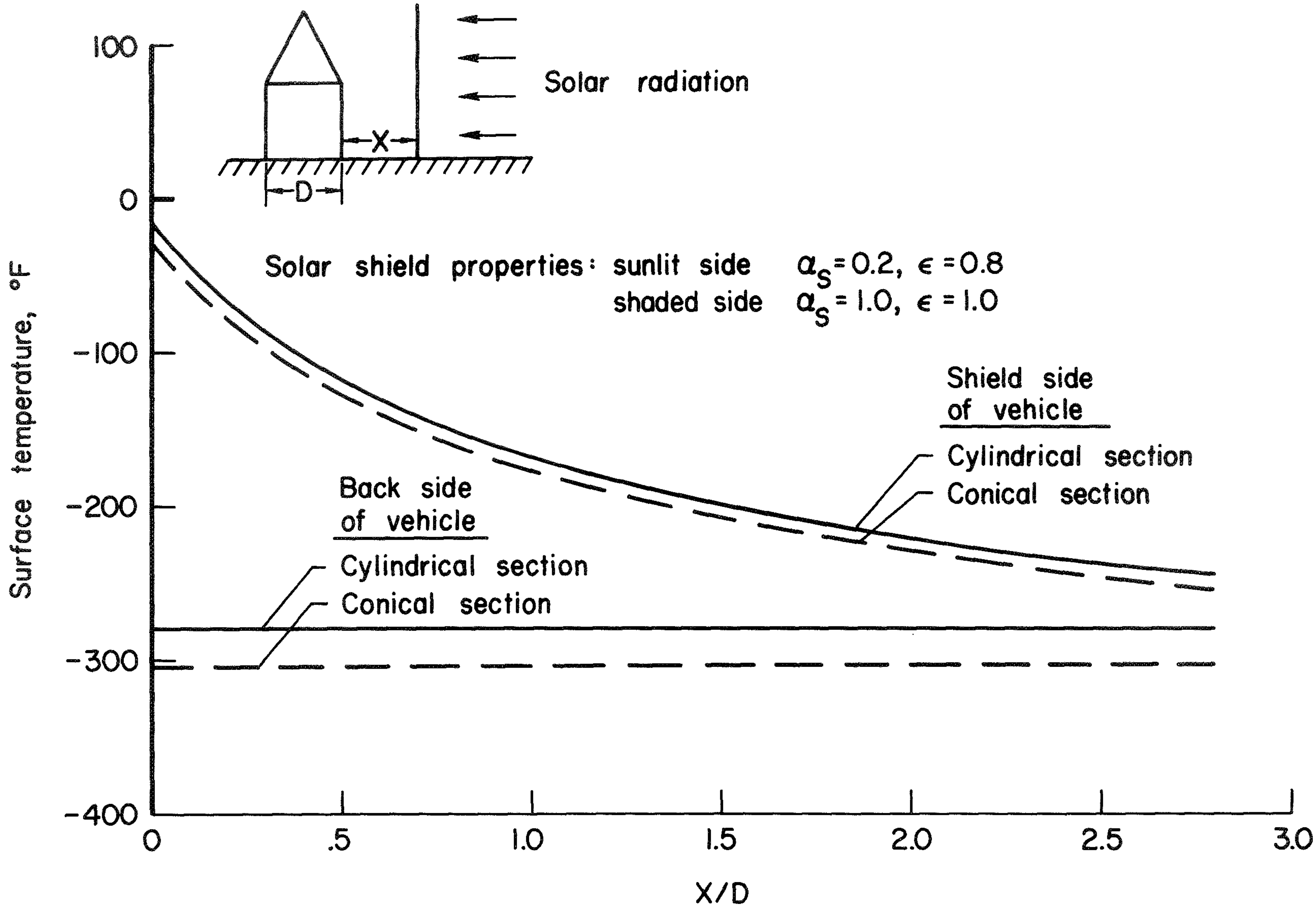


Figure 8. - Effectiveness of a solar-radiation shield when the vehicle is illuminated from the side; conductance and internal heat both zero.

N66 32950

DISCUSSION OF THE OKLAHOMA STATE UNIVERSITY ACTIVE TEMPERATURE CONTROL SYSTEM

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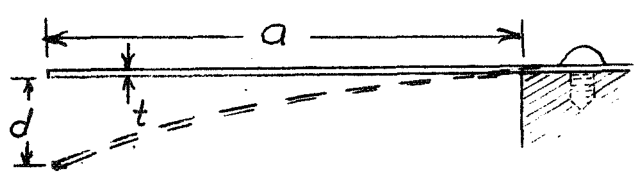
This discussion summarizes the work done under NASA Grant NsG-454 at Oklahoma State University as described in the Semi-Annual Report "Spacecraft Temperature Control by Thermostatic Fins" by J. A. Wiebelt and J. F. Parmer. A very interesting and potentially useful active temperature control device is being developed (see Figure 1). It consists of pairs of bimetallic strips mounted back-to-back. At a specified temperature (say $100^{\circ}F$) the bimetallic strips are straight and perpendicular to the mounting surface. If the mounting surface, which represents the skin or radiating surface of a spacecraft, is painted white and the bimetallic surfaces are coated with a highly reflecting material such as evaporated aluminum, the vertical position would have the highest effective emittance. When the temperature is less than the temperature corresponding to the vertical position the adjacent bimetallic elements bend away from each other and reduce the effective emittance of the total surface.

The bimetallic material chosen for analysis and fabrication of an experimental unit is Truflex P-675R manufactured by the Metals and Controls Division of Texas Instruments, Inc. The deflection equation is as follows:



$$d = 113 \times 10^{-7} (\Delta T) \frac{a^2}{t}$$

From Figure 2 it can be seen that sunlight incident on the grooves may strike the base directly or it may be reflected from the walls one or more times before striking the base. Each time a ray of sunlight strikes the walls, part of its energy is absorbed and the rest is reflected specularly. When a ray strikes the base, part of the energy is absorbed and the rest is reflected diffusively. Part of the diffusively reflected energy escapes directly through the groove opening and the rest is partially absorbed and reflected specularly by one or more collisions with the groove walls before escaping. Obviously, a great deal of effort is required to trace the history of each ray of sunlight in order to arrive at a value of effective solar reflectance. The details of this analysis are described in the semi-annual report. The results are summarized in Figures 3, 4, and 5. (All Figures are taken directly from the semi-annual report) The curves are plotted showing the ratio of effective reflectance to base reflectance as a function of solar polar angle, with wall reflectance as a parameter. As one might expect the effective reflectance falls off rapidly as the solar polar angle approaches 90°, increases with increase in wall reflectance and decreases with increase in a/b, a measure of the effective depth of the grooves.

Wavelength Dependence of Effective Reflectance

Since the effective reflectance is a function of both wall and base reflectances, one specular and the other diffuse, it can be expected to vary with wavelength. To obtain the effective solar reflectance it is necessary to integrate over the solar spectrum the effective spectral reflectances weighted by the relative solar intensity at each wavelength interval. Figure 6 shows the reflectances of evaporated aluminum, a white paint, and the effective reflectances of the combination as a function of wavelength. The integrated values are shown in Figure 7.

Heat Dissipation Characteristics

An analysis was made to determine the net heat flux from the surface. The following assumptions were made:

1. Fins are vertical and specular;
2. The base is horizontal and diffuse;
3. Fins are infinitely long;
4. Conduction between fins and base is negligible;
5. Fin and base temperatures are equal;
6. Emittance and solar absorptance values for the groove are determined for the vertical position and are assumed to be independent of temperature;
7. Emittance and solar absorptance of the slit = 1.0 and are independent of temperature; and
8. The relative surface areas of grooves and slits are determined from the deflection equation for the fin material.

In the analysis of solar reflectance, only the vertical position for the fins was considered. If the deflection of the fins from the vertical position is small, the values of emittance and solar absorptance are reasonably accurate for positions other than the vertical position. As the fins separate slightly, a V-shaped slit opens between adjacent fins. For small deflections this slit can be assumed to be optically black even though the surfaces are highly reflective. As the deflection increases significantly from the vertical position the error increases appreciably. The groove takes on the properties of a cavity and the slit becomes more reflective.

In Figures 8 and 9, net heat flux is plotted as a function of solar polar angle, with fin temperature as a parameter. The solutions for temperatures near 110°F, the vertical position, are reasonably accurate. Figures 10 and 11 show surface temperature as a function of solar polar angle for a fixed internal heat dissipation of 37.5 BTU/Hr.-Ft.^{2}. Again the curves are reasonably accurate for temperatures near 110°F.

These curves indicate that the surface temperature is constant over a fairly wide range of solar polar angles. This is the result of a decrease in reflectance or an increase in absorptance with solar polar angle as shown in Figure 7 combined with the decrease in incident solar flux with solar polar angle.

Future Plans (Proposed)

Oklahoma State University has proposed to extend the analysis by: (1) Removing the restriction that the fins be vertical in determining effective reflectance, (2) considering fins of finite length, and (3) allowing for conduction between the fins and the base. In addition, an experimental unit would be fabricated and tested under solar simulation. Goddard would provide the solar simulation facilities.

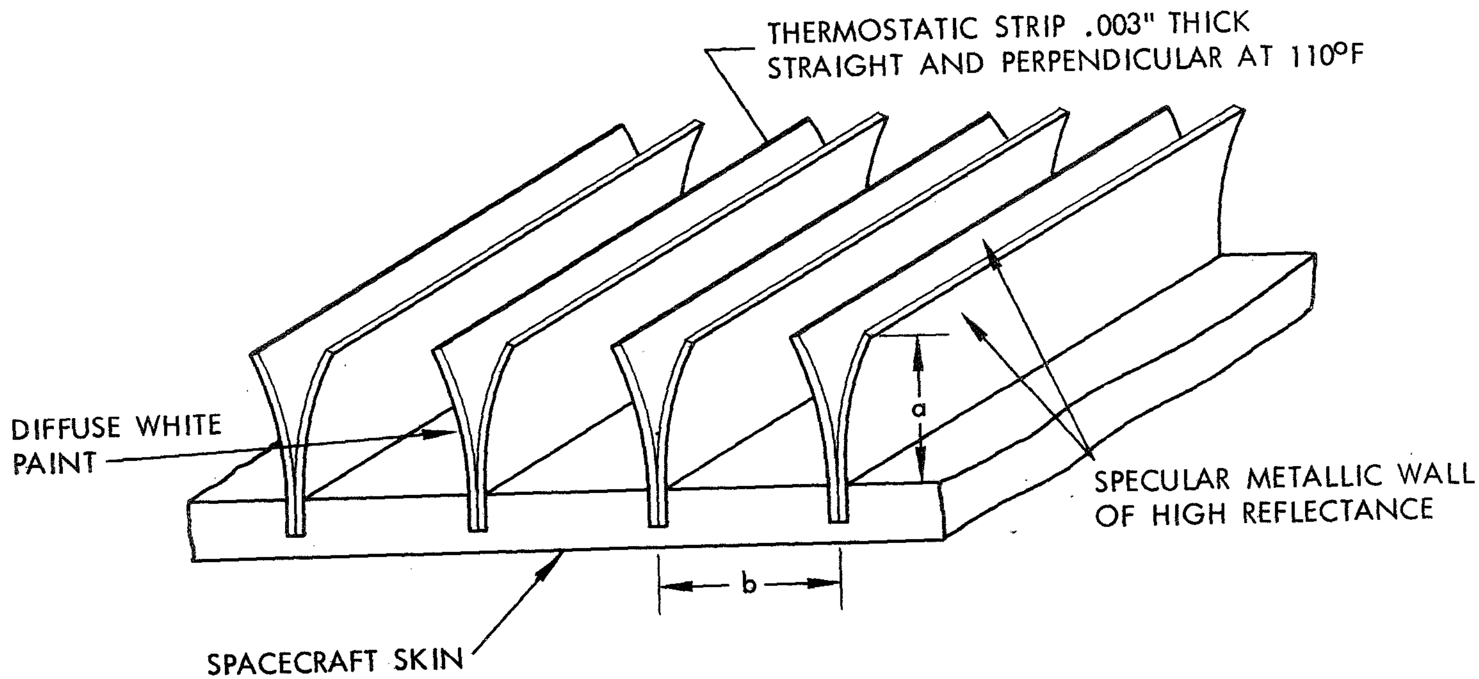


Figure 1. Thermostat surface

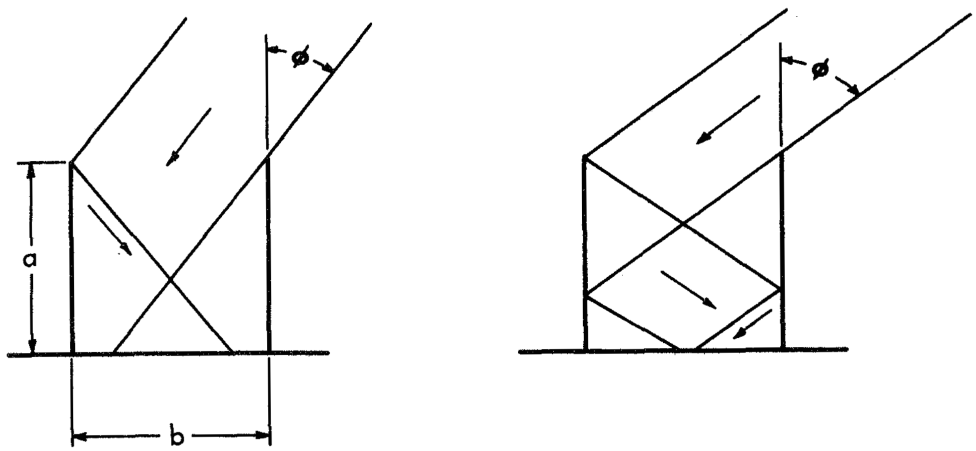


Figure 2. Multiple reflections of solar illumination

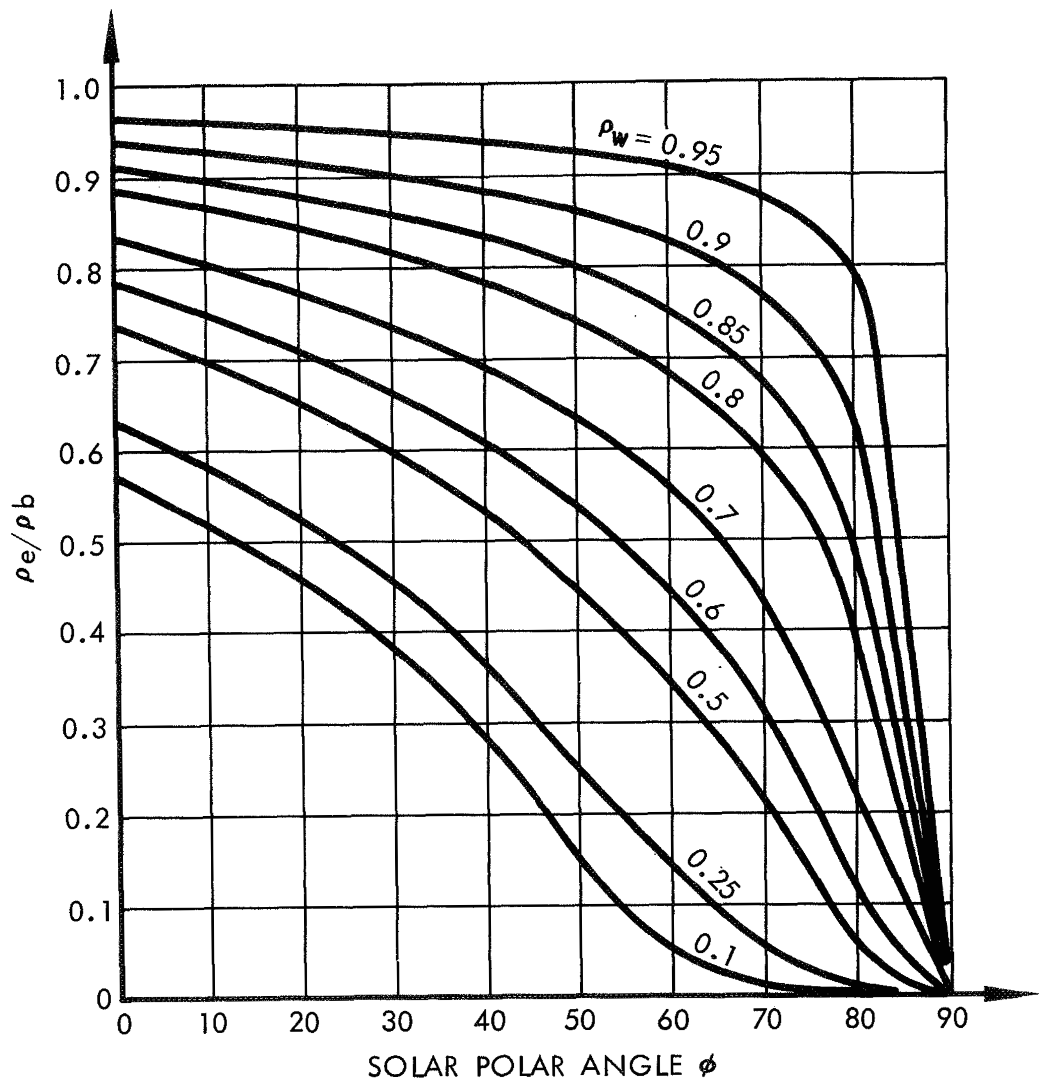


Figure 3. Effective reflectance ratio, a/b = 2/3

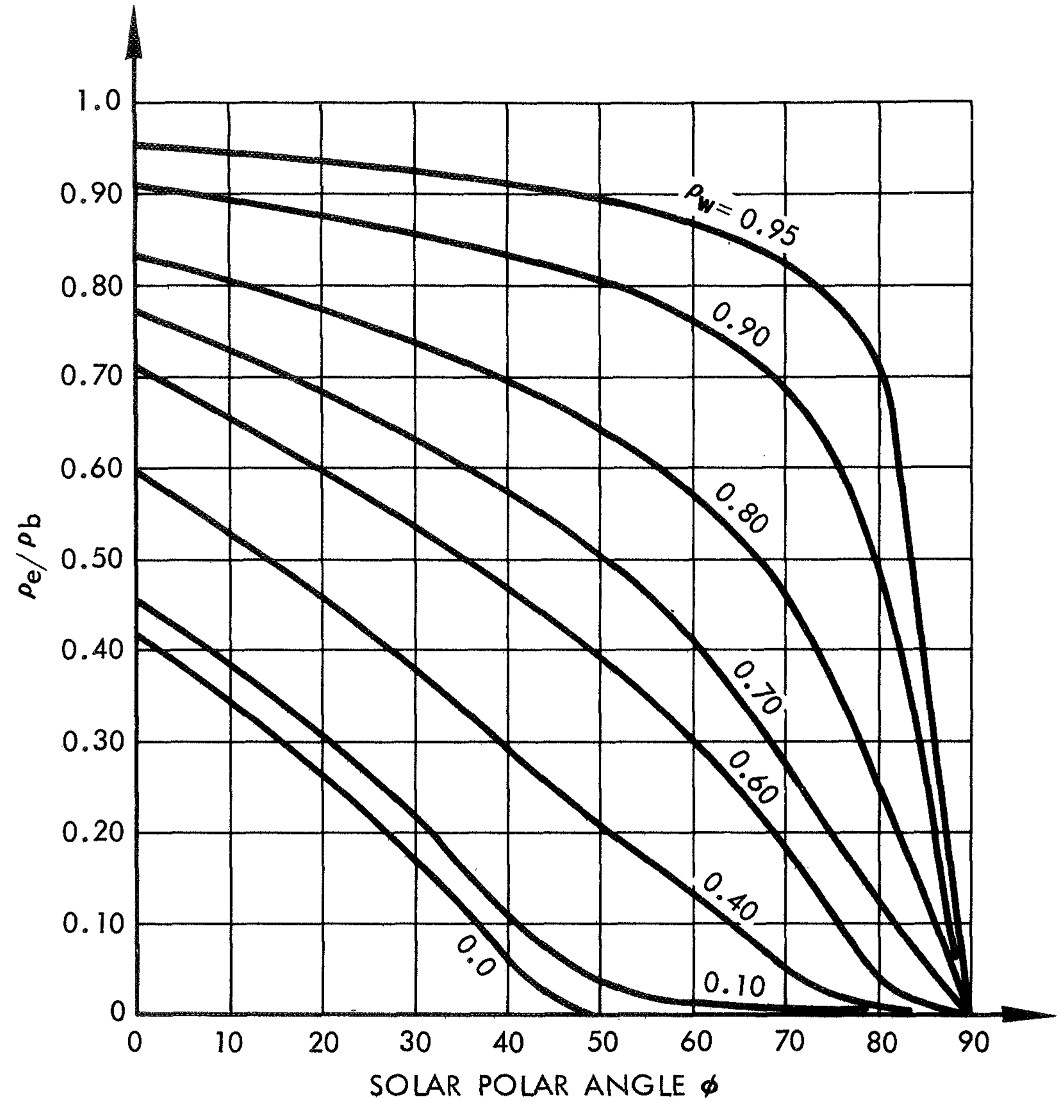


Figure 4. Effective reflectance ratio, a/b = 1

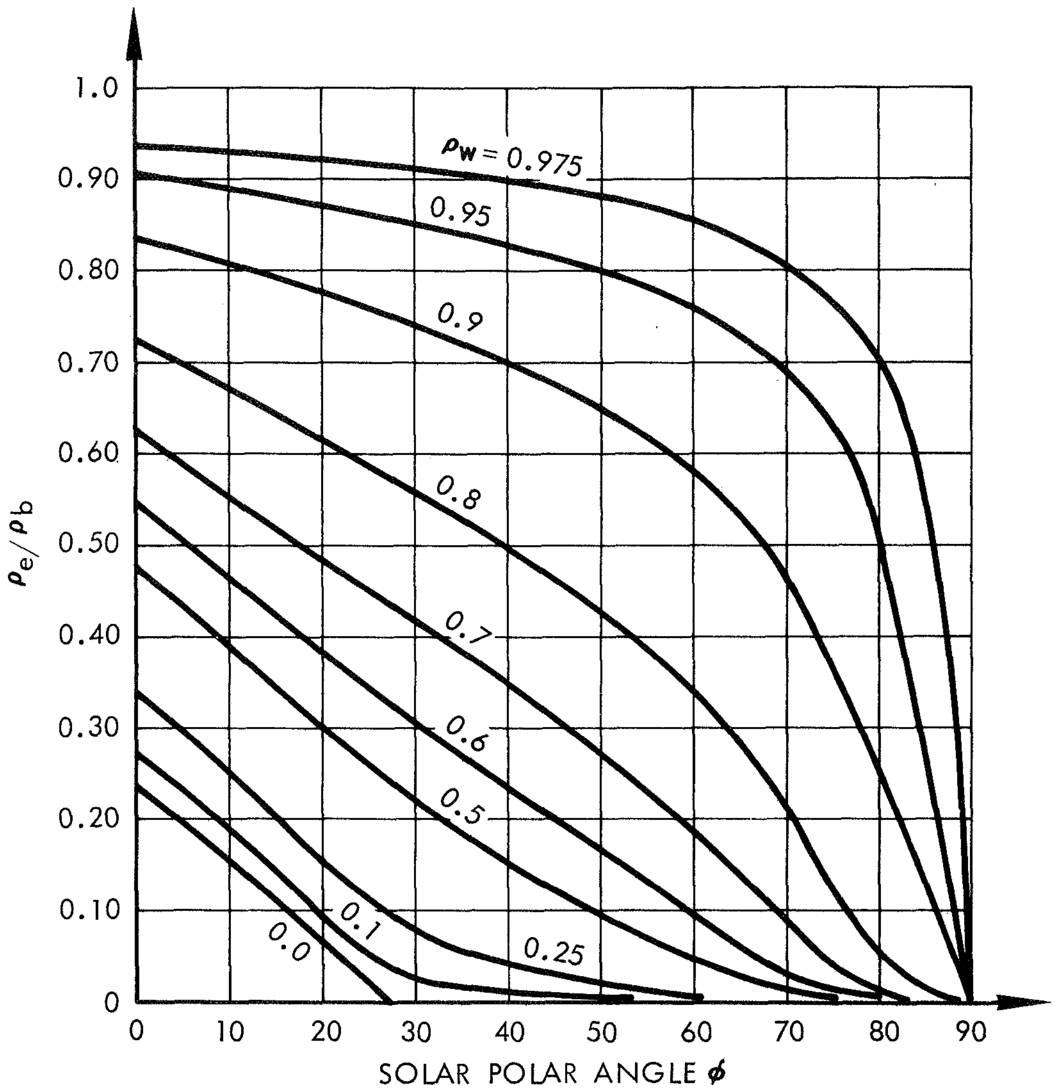


Figure 5. Effective reflectance ratio, a/b = 2

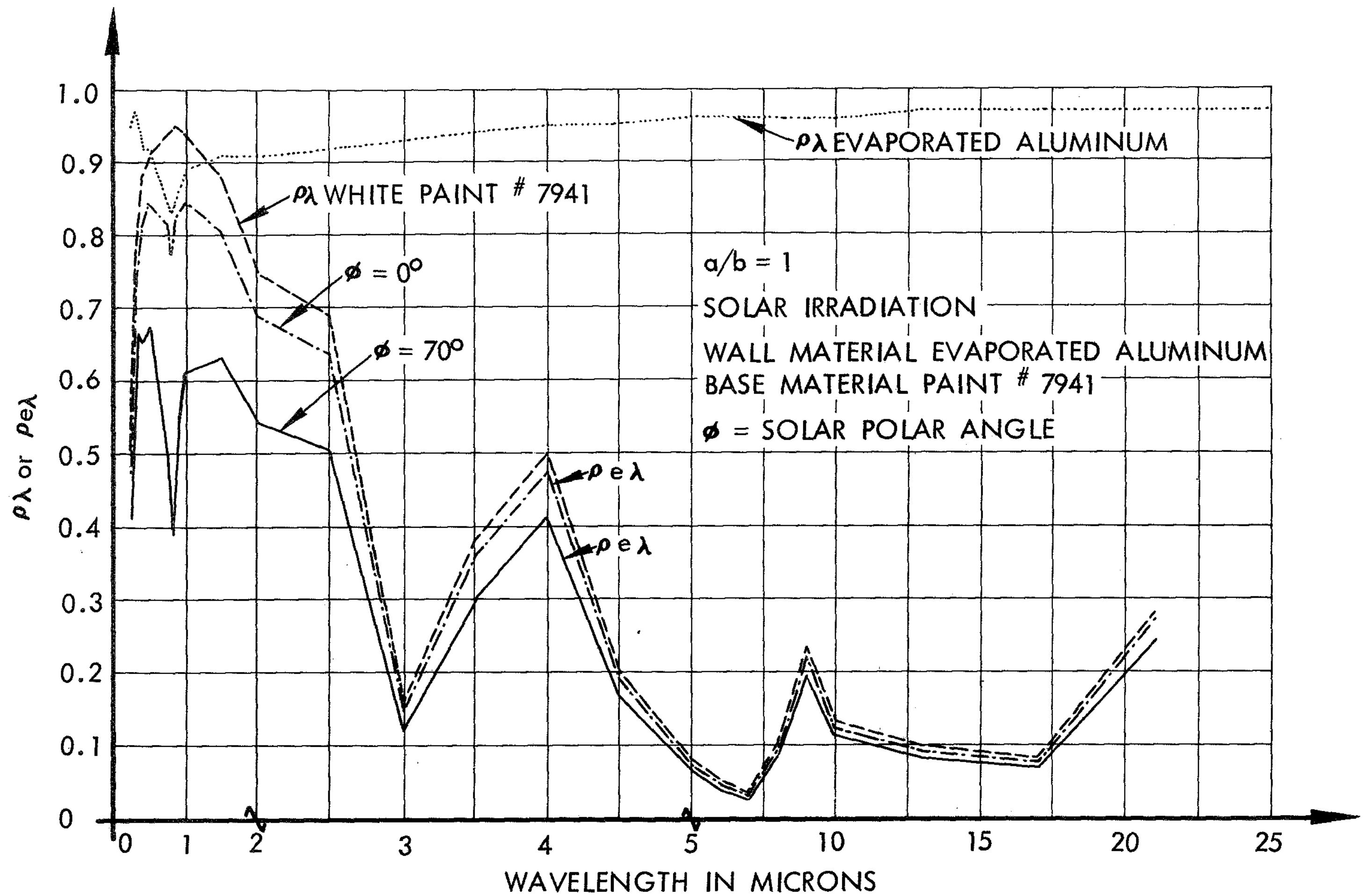


Figure 6. Effective monochromatic reflectance

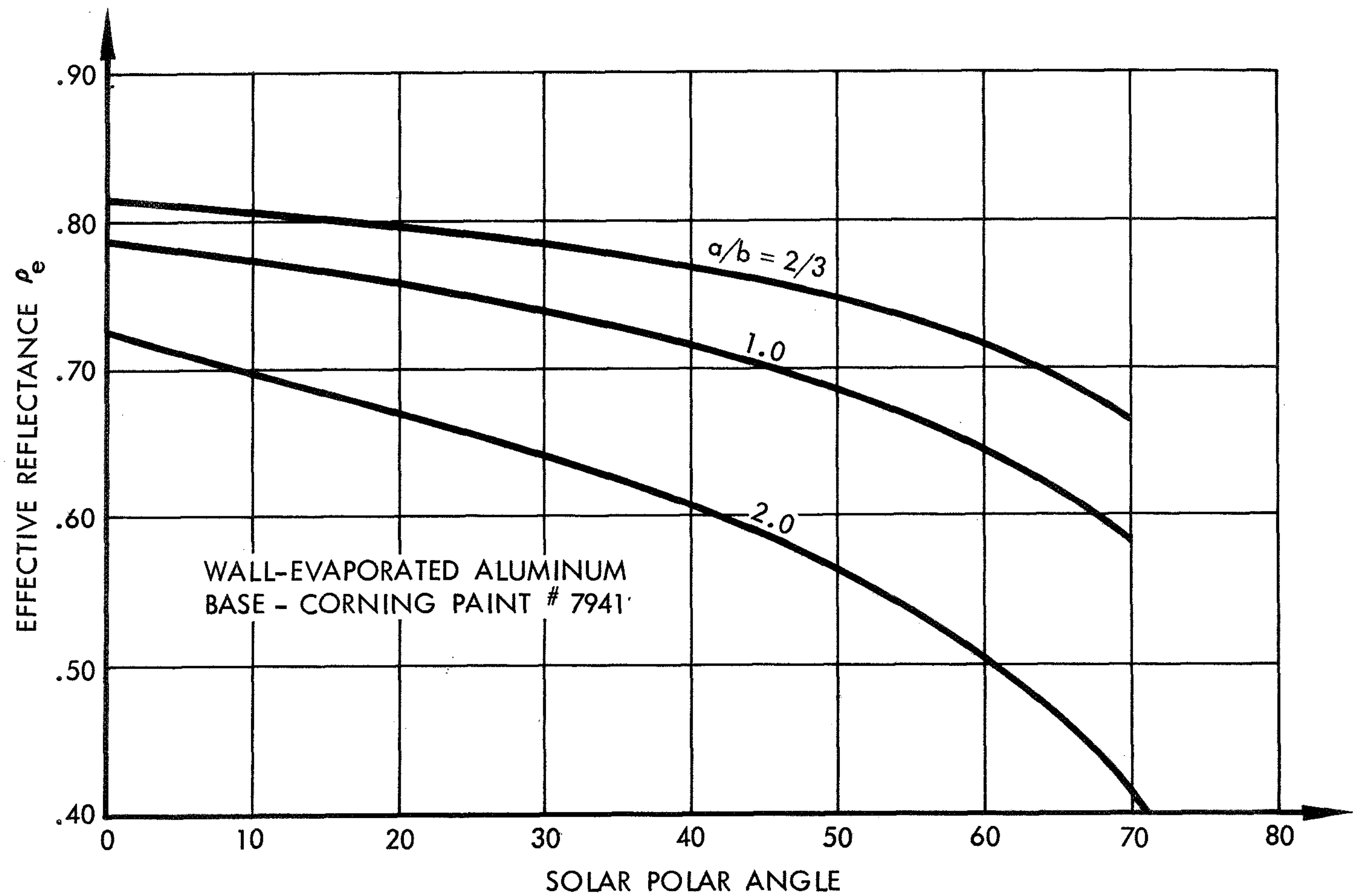


Figure 7. Effective reflectance

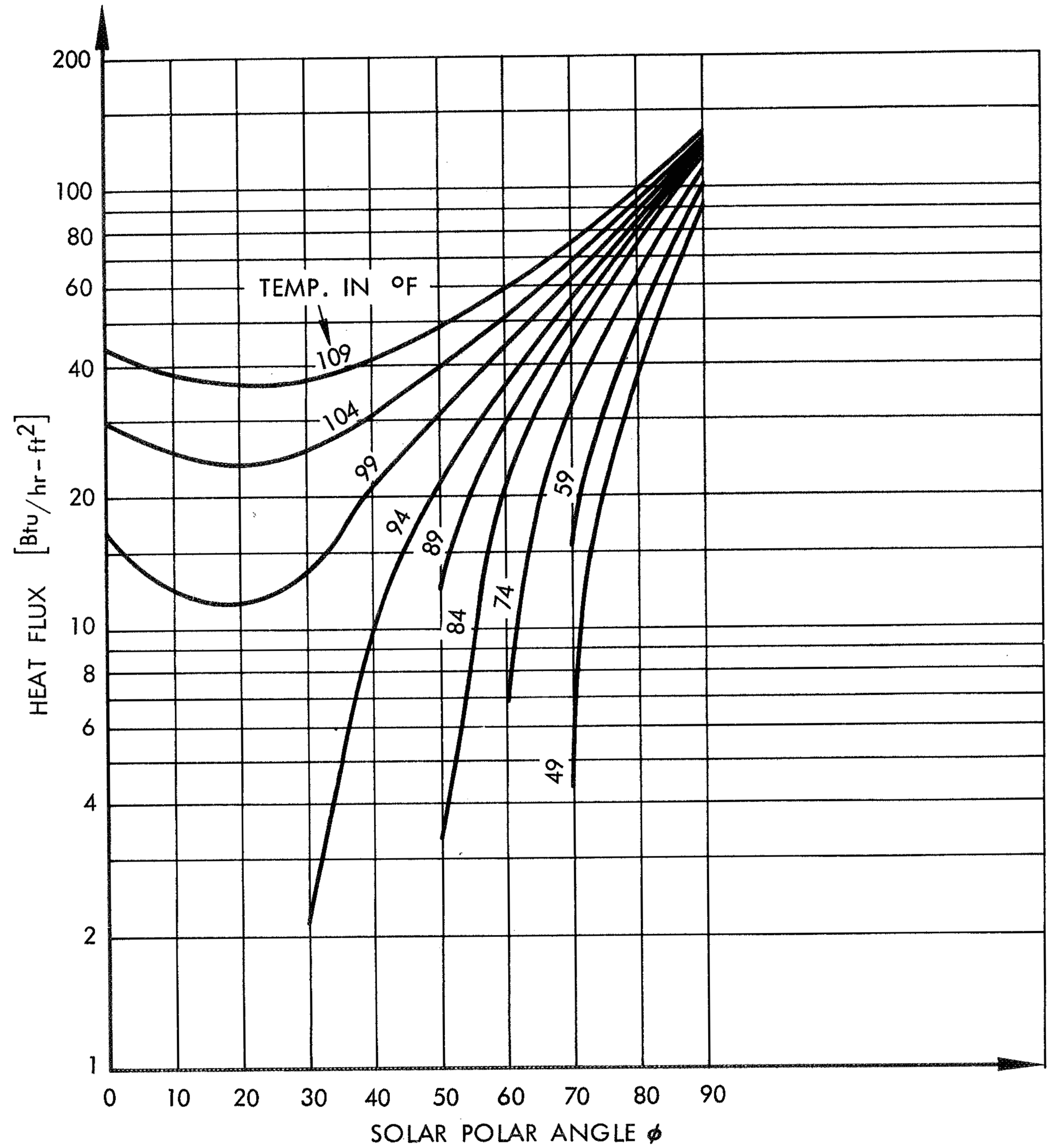


Figure 8. Equilibrium temperatures for a .75" x .75" thermostat surface

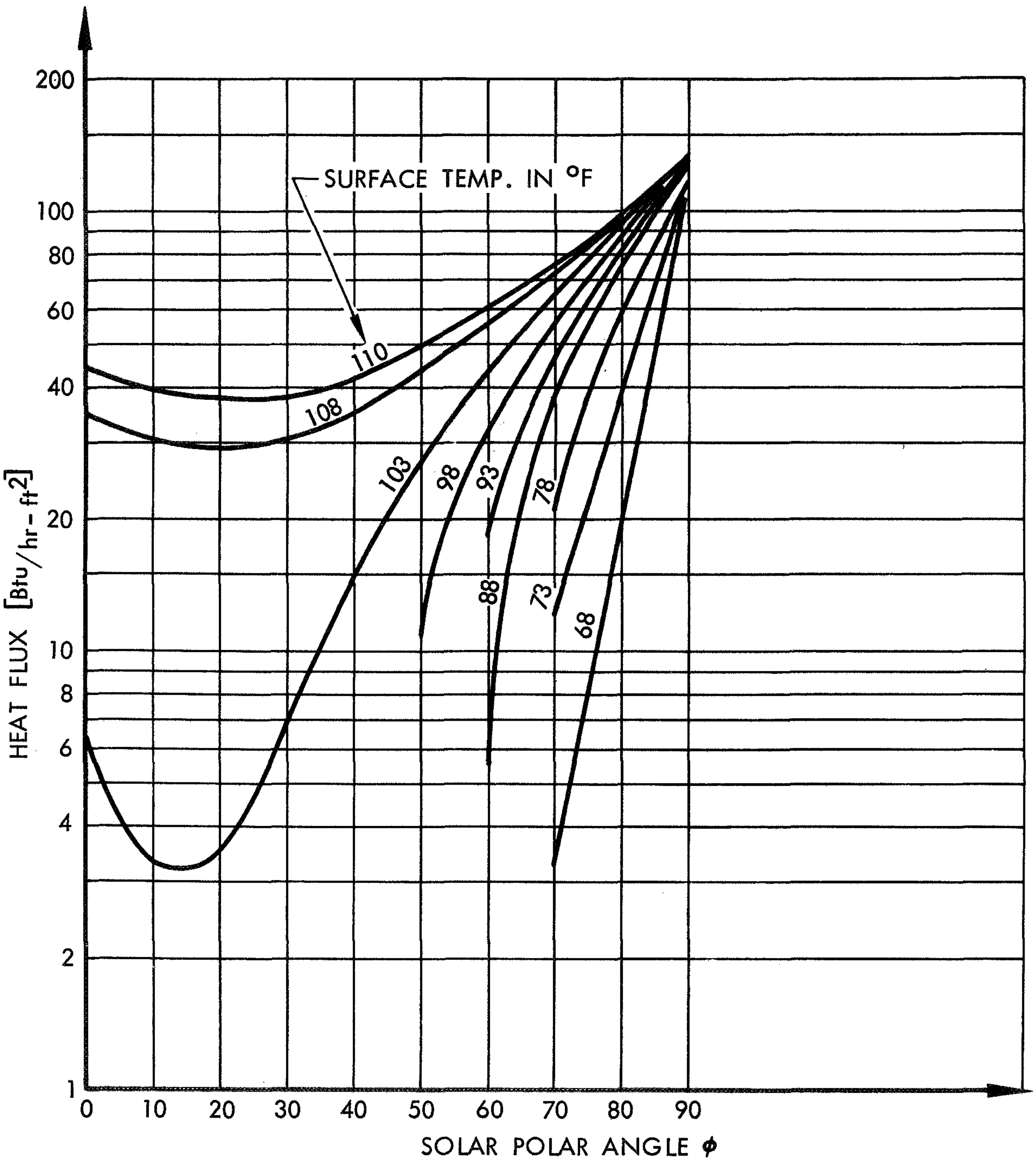


Figure 9. Equilibrium temperatures for a 2" x 2" thermostat surface

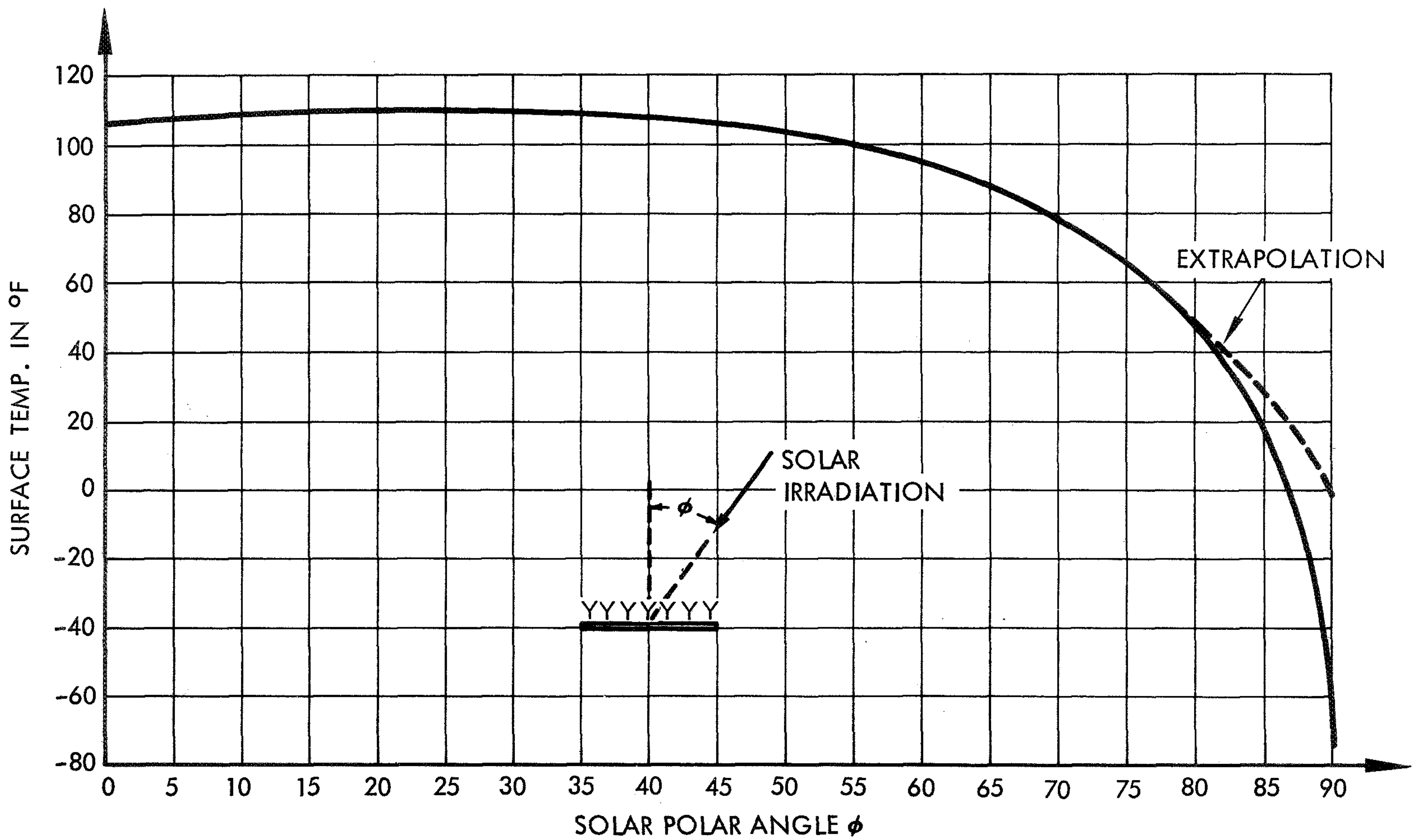


Figure 10. Surface temperature of a thermostat surface with fin height (a) of 0.75 inches, fin spacing (b) of 0.75 inches with internal heat generation of the spacecraft such that 37.5 Btu/hr-ft^2 must be radiated to space.

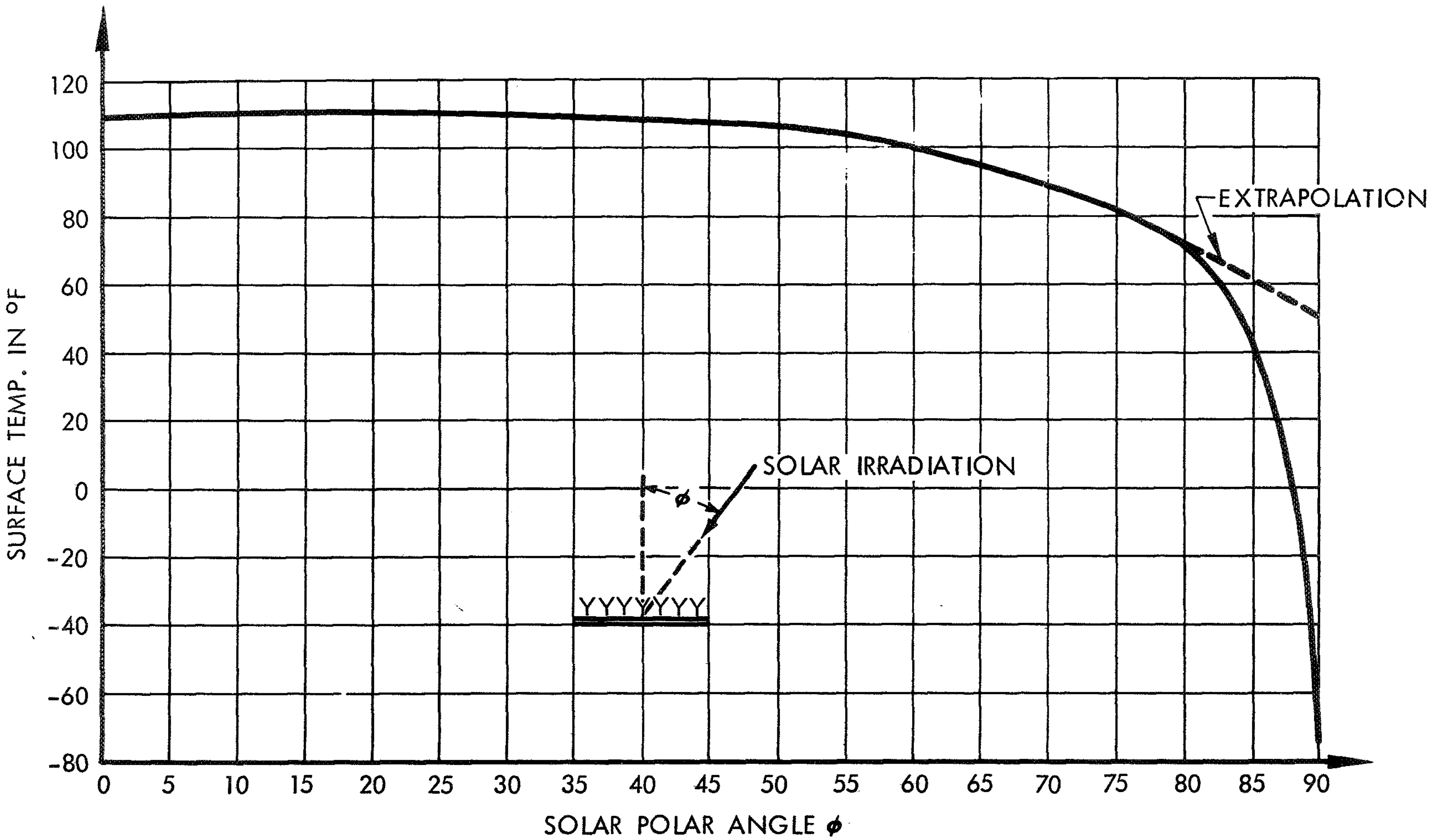


Figure 11. Surface temperature of a thermostat surface with fin height (a) of 2.0 inches, fin spacing (b) 2.0 inches with internal heat generation of the spacecraft such that 37.5 Btu/hr-ft^2 must be radiated to space.

N66 32951

CONTROLLABLE α/ϵ RATIO SURFACES FOR TEMPERATURE REGULATION OF SPACECRAFT

By William J. O'Sullivan, Jr.

ABSTRACT

The research and development program on Controllable α/ϵ Ratio Surfaces for Temperature Regulation of Spacecraft of the Space Vehicle Branch, Applied Materials and Physics Division, NASA, Langley Research Center is described in its entirety. There is given the technical background defining the problem, the possible solutions found from application of electromagnetic and quantum mechanics theory, the classification of the potential solutions by method of operation into thermotropic and nonthermotropic types, the experimental research program and its division into Phase I aimed at obtaining basic data on the phenomena and Phase II aimed at engineering development of the most promising methods, details of operation of the change in transparency method representing the most successful thermotropic method found to date, and details of the Electroluminescence method representing the most successful nonthermotropic method found to date.

CONTROLLABLE α/ϵ RATIO SURFACES FOR TEMPERATURE REGULATION OF SPACECRAFT

By William J. O'Sullivan, Jr.

1. Introduction.- The Langley Research Center's Space Vehicle Branch is engaged in a research program consisting of the devising, exploratory investigation, and development of what it calls Controllable α/ϵ Ratio Surfaces for Temperature Regulation of Spacecraft. The optical property α is the absorptivity of the spacecraft's surface to solar radiation, and ϵ is its emissivity to thermal radiation. This paper briefly describes the concepts of these controllable α/ϵ ratio surfaces and the progress to date in their development.

2. Technical background.- It is probably now a matter of common knowledge that in the hard vacuum of space a spacecraft can receive and lose heat only by the process of radiative heat exchange. A general proof has been developed (ref. 1) that the temperature of a spacecraft of any given configuration whose exterior surfaces have fixed ratios of α/ϵ, depends only on three things: (1) The location of the spacecraft relative to external sources of heat, that is, the sun and any nearby planetary body like the earth or moon. (2) The attitude of the spacecraft relative to these external heat sources. (3) The rate of heat release inside the spacecraft from chemical or nuclear energy sources. There is no possible orbit or spacecraft mission of any consequence in which all three of these factors can remain constant. Depending on the particular spacecraft mission and

orbit, designers have resorted to various fixes to hold the temperature of the spacecraft within permissible limits, but always at some penalty. Spinning has been used to expose all sides equally to the incident radiation, but this is incompatible with most missions. Mechanical shutters have been used to change the α/ϵ ratio of the exterior surfaces, but these are heavy and degrade reliability. Heat has been released internally and thermal insulation used, but this is heavy and consumes precious onboard power. Fixed orientation toward the sun has been used, but this requires a control system and continuous expenditure of mass to operate control jets. So many spacecraft designers have voiced their wish for a surface that would change its α/ϵ ratio spontaneously or on command as needed to hold the spacecraft's temperature constant that this imaginary surface has come to be known as a chameleon surface. The Langley Space Vehicle Branch's research program on controllable α/ϵ ratio surfaces is a deliberate attempt to devise and develop such surfaces that are practical and reliable.

3. Development of basic concepts.- The first step carried out in the Langley Space Vehicle Branch's program was to discover ways of getting changeable α/ϵ ratio surfaces. To find this out a search (ref. 2) was made deep into theory, especially electromagnetic theory and quantum mechanics, to try to understand what basically caused the α and ϵ of surfaces to be what they are, and whether there was any relationship between them. It was found that theory was very speculative and poorly developed in this area, but the phenomena seem to depend in large measure

upon the electron population within the material, with the result that anything that can change the electron population should change the α/ϵ ratio. With this clue as a guide, search was then made for materials and phenomena by which change in the electron population could be accomplished, and a surprising number were found (ref. 2). Change in electron population is to be understood in the broad sense of meaning not simply the number of electrons per unit volume, but also their degree of mobility or lack thereof, and their position within the molecular and crystal structure. Phenomena identified in this manner as being potentially capable of yielding controllable α/ϵ ratio surfaces were:

1. Electroluminescence
2. Semiconductors
3. Photoemissivity
4. Photoconductivity
5. Photovoltaic effect
6. Change in electrical resistance with temperature
7. Magneto-optical rotation
8. Polarization
9. Chemical indicators
10. Change in transparency

In addition, two mechanical-optical methods that will not here be discussed were found.

4. Classification by method of operation.- The ten aforementioned phenomena by means of which surfaces can be made whose α/ϵ ratio can be changed can be classified into two major groups with respect to how they would work. The previously mentioned study showing that the temperature of a spacecraft is a function of its α/ϵ ratio immediately indicates that change in α/ϵ ratio has to be controlled by the temperature of the spacecraft's surface. This immediately sorts the changeable α/ϵ ratio surfaces into two categories. (1) Those whose change in α/ϵ ratio is produced directly by a change in their temperature, for which reason they might be called thermotropic surfaces. (2) Those whose change in α/ϵ ratio is produced by some means other than temperature, and for distinction may be called nonthermotropic.

The thermotropic type surface, when subject to solar radiation and at a temperature below its control temperature, has a high α/ϵ ratio and therefore absorbs more heat than it radiates so that its temperature rises. When it reaches its control temperature, it changes to a low α/ϵ ratio and emits more heat than it absorbs, and therefore its temperature starts to decrease. But this causes it to change back to a high α/ϵ ratio, and thus its temperature is automatically controlled to the temperature at which it undergoes its change from high to low α/ϵ ratio. The thermotropic type surface, since it spontaneously changes, probably represents the acme of reliability.

The nonthermotropic type surfaces operate with a thermostate or other temperature sensor attached to them to measure their temperature. When they are below their control temperature they have a high α/ϵ ratio and absorb more heat from the incident solar radiation than they emit, with the result that their temperature rises. When they reach their control temperature as detected by the temperature sensor, the temperature sensor commands the activation unit to change the α/ϵ ratio of the surface to a low value. The surface then emits more heat than it absorbs and its temperature starts to decrease. But this causes the temperature sensor to command the activator unit to change the surface back to a high α/ϵ ratio, and thus its temperature is automatically controlled to the control temperature at which the temperature sensor is set. Thus, by simply changing the control temperature setting of the temperature sensor the nonthermotropic type controllable α/ϵ ratio surface can be made to automatically control its temperature to any desired value over a range of temperatures. But this advantage is achieved at some increase in complexity, and hence decrease in reliability, over the thermotropic type surface.

It is to be noted that in both the thermotropic and the nonthermotropic type surfaces herein discussed, the change in α/ϵ ratio is a distinct step change as contrasted to the gradual change of α/ϵ with temperature that has been noted in some materials and proposed for use on spacecraft by others.

Of the preceding list of ten phenomena giving such controllable α/ϵ ratio surfaces, phenomena 1 through 5 and 7 through 9 can operate as nonthermotropic surfaces; phenomena 9 as either a thermotropic or nonthermotropic surface; and phenomena 6 and 10 as thermotropic surfaces.

5. Experimental research and development program.- The experimental research and development program worked out jointly between NASA Headquarters and the Langley Space Vehicle Branch is divided into two sequential phases.

Phase I is aimed at providing experimental measurements of the magnitude of the change in α/ϵ ratio that can be produced by each of the aforementioned ten phenomena and the temperature range over which they can be made to occur. On this basis the phenomena are being sorted to determine which are the most promising and worthy of being pursued further in Phase II, as well as to provide the foundation data needed for Phase II. Phase I is being carried out partly in-house and partly out-of-house by a Research Grant to Georgetown University. The division of the research between these two places is based upon the capabilities of the technical personnel at each location. Since suitable experimental apparatus for satisfactorily performing this research was not in existence, such had to first be designed, constructed, and brought into operation before the measurements could be made. This has been done and successful measurements have already been made on a number of the phenomena. This Phase I is still in progress.

Phase II is aimed at carrying out engineering development on the most promising phenomena resulting from Phase I. By engineering development is meant the host of problems associated with converting a device from a laboratory model into a practical thing for use on spacecraft, and includes such things as fabrication methods, quality and reliability controls and standards, and the like. Already, several of the phenomena have just reached this stage.

6. The change in transparency method.- The first of the thermotropic type surfaces to emerge from Phase I as definitely worthy of proceeding into Phase II is based on the change of transparency phenomenon which is number 10 of the preceding list. Its Phase I investigation was carried out at the Langley Space Vehicle Branch.

As shown schematically in cross section in figure 1, it consists of a layer of changeable transparency material that forms the exterior surface of the spacecraft, behind which there is a mirror coating, which in turn is backed up by an opaque rear layer that forms the skin of the spacecraft. The changeable material layer consists of a transparent matrix material, such as a plastic, in which there exists a dispersed phase shown schematically as little spherules. When the temperature of the spacecraft's surface is below the control temperature of the coating, the dispersed material is opaque and the coating is in the absorbing condition, having a high α/ϵ ratio so that it absorbs more energy from the incident sunlight than it emits, and its temperature rises. When it reaches the control temperature,

the dispersed material undergoes a phase change and becomes transparent, so that the incident light passes through the changeable transparency material, is reflected by the mirror backing, passes again through the changeable transparency material, and then out through its front face, so that the coating in effect has a low α/ϵ ratio, and its temperature tends to decrease. But if its temperature decreases, the dispersed material changes back to its opaque state and the high α/ϵ ratio is restored. The coating thus automatically regulates itself to the control temperature which is the temperature at which the change of transparency occurs in the dispersed material.

An almost infinite number of materials can be used as the dispersed material giving a vast range of control temperatures. The change from absorbing to reflecting condition occurs over a very narrow range of temperature depending on the purity of the material used, being in some cases tried only a fraction of a degree.

7. <u>The electroluminescence method.</u>- The first of the nonthermotropic type surfaces to emerge from Phase I as definitely worthy of proceeding into Phase II is based on the electroluminescence phenomenon which is number 1 of the preceding list. Its Phase I investigation was carried out at Georgetown University.

As shown schematically in cross section in figure 2, it consists of a layer of electroluminescent material having a transparent electric conductor on its front face and an ordinary electric conductor on its back,

thus forming a capacitor in which the electroluminescent material is the dielectric. This three-layer sandwich is mounted on the exterior of the spacecraft's wall with the transparent electric conducting layer facing outward. Mounted on, or imbedded in, the inner side of the spacecraft's wall is the temperature sensor. When the spacecraft's wall is below the control temperature, the electroluminescent material is in its high α/ϵ ratio state so that it absorbs more heat from the incident solar radiation than it emits, and therefore the temperature of the spacecraft's wall rises. When it reaches the control temperature, as detected by the temperature sensor, the temperature sensor actuates an electric switch, sending a command signal to the activator unit which the applies across the electroluminescent material the electrical potential necessary to change it to its low α/ϵ ratio state. The electroluminescent material then absorbs less heat from the incident solar radiation than it emits so that the temperature of the spacecraft's wall starts to decrease. But any decrease is immediately detected by the temperature sensor which signals the activator unit to change the potential on the electroluminescent material so as to change it back to the high α/ϵ ratio state. Thus the temperature of the spacecraft's wall is automatically regulated to the value of the control temperature set into the temperature sensor. On some electroluminescent materials tried the electrical power required to hold the electroluminescent material in the changed α/ϵ ratio state is less than 0.2 watt per square foot of surface, and thus the power consumption is extremely modest.

This, together with the very thin electroluminescent layer required, of the order of a fiftieth of an inch, which makes the surface very lightweight, indicates that the electroluminescence method is practical.

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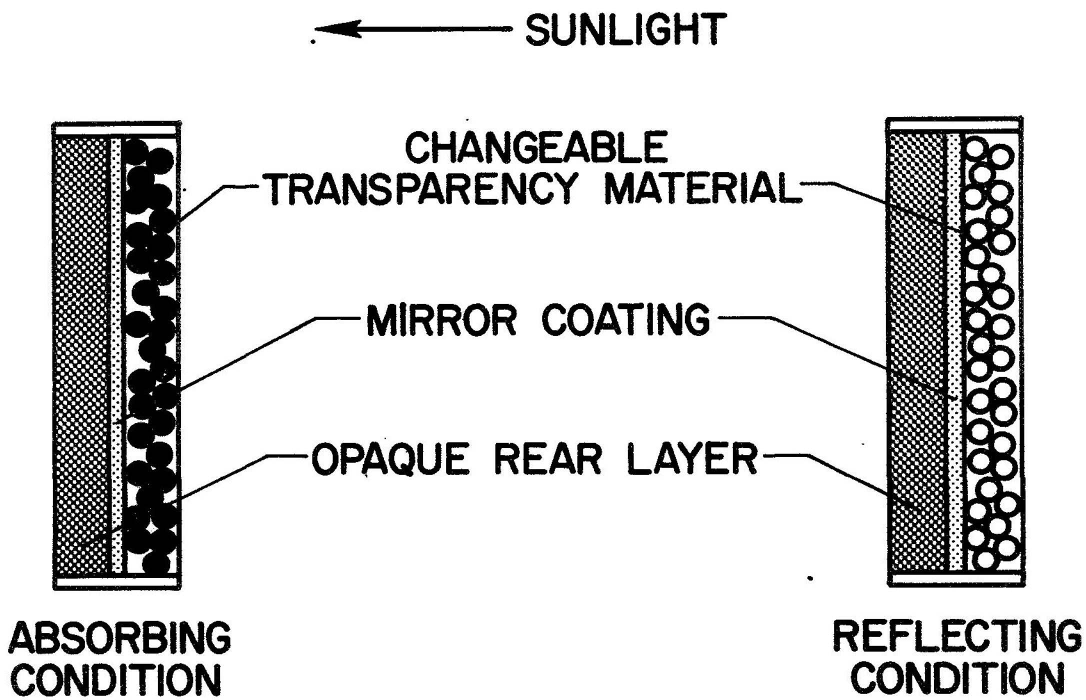
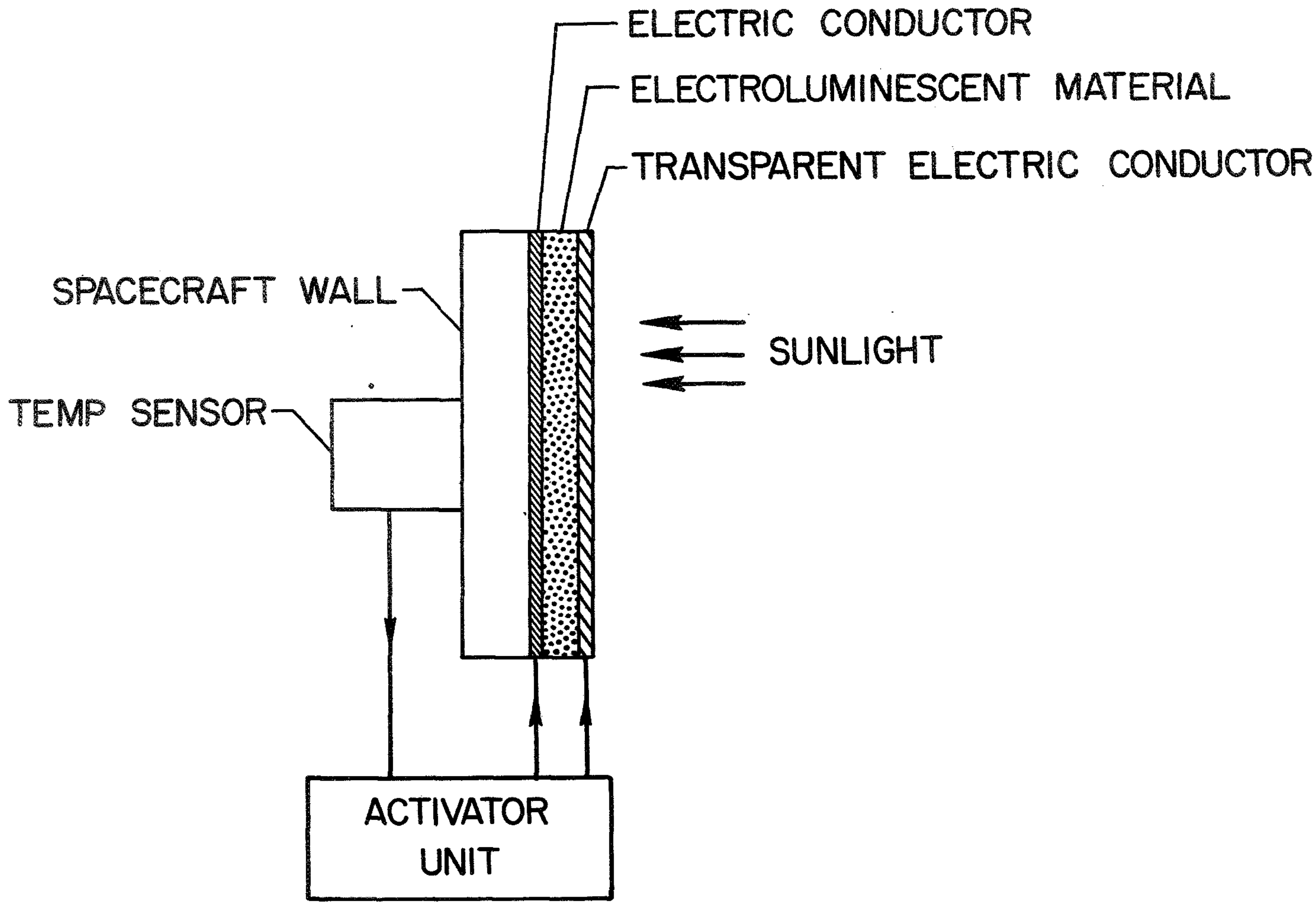
2. NASA-Langley Memo. for Assoc. Dir. (Atten. E. C. Draley), Nov. 22, 1961, by W. J. O'Sullivan. Proposal of a Potentially Revolutionary Means of Accomplishing the Thermal Balance of Spacecraft and Request for Decision on Means of Pursuing This Idea.

SCHEMATIC CROSS-SECTION OF CHANGEABLE TRANSPARENCY TYPE OF CONTROLLABLE α/ϵ RATIO SURFACE



SCHEMATIC CROSS-SECTION OF ELECTROLUMINESCENT TYPE OF CHANGEABLE α/ϵ RATIO SURFACE



PHASE CHANGE IN POLYMERIC SYSTEMS FOR ACTIVE THERMAL CONTROL

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NASA-Langley Research Center

INTRODUCTION

Many materials undergo significant changes in their optical properties when they undergo first order phase changes. This is particularly true of polycrystalline organic compounds. An exploratory research program has been initiated in the Spacecraft Materials Section of the Space Vehicle Branch to study this phenomena of phase change for possible use as a means of automatically controlling (without moving parts) the optical properties and hence the temperature of a spacecraft's surface. Very little has been done in the past to explore or utilize this phenomena for temperature control. One brief description of some earlier work on using phase change for temperature control was disclosed in the early 1950's by G. W. Kuehl (reference 1) who investigated a series of compounds which changed their optical properties with temperature or light intensity. Since these compounds also changed their physical state, they generally had to be contained between the walls of multi-layer glass sheets.

The model systems being studied at Langley are self-contained coatings which can be made to vary their absorptance of solar radiation at pre-designed temperature levels. This variation in solar absorptance is due

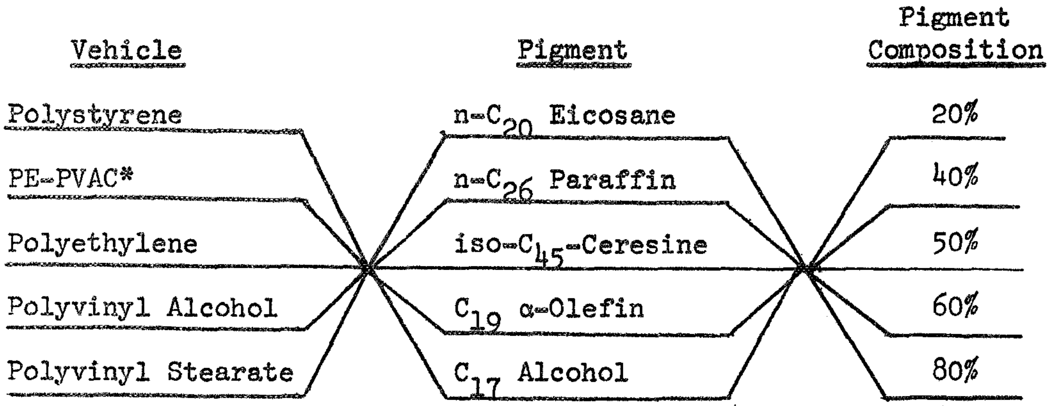
to a phase change within a part of the coating itself. At the phase transition temperature the coating's optical properties change from a diffusely reflecting material to a transparent material. This change occurs at the melting point of polycrystalline pigment which is dispersed throughout the transparent coating vehicle or matrix.

Below its melting temperature, the dispersion of the polycrystalline pigment in a transparent vehicle produces numerous complex reflecting surfaces which cause the coating system to behave like a typical diffuse reflector of light. Above its melting temperature, the polycrystalline pigment is amorphous and is effectively soluble in the transparent vehicle which is still solid. This results in a transparent coating system which permits light to pass through the coating to a mirror substrate which reflects the light back essentially unabsorbed (figure 1). The coating system with its mirror substrate has a higher solar absorptance when the coating is a diffused reflector than when it is in the transparent condition. This is due to the greater number of reflecting surfaces which exist at the lower temperature. When the pigment exists in the polycrystalline state, these internal reflecting surfaces effectively increase the length of the incident light path in the coating and hence increase the overall absorption.

Experimental Coatings

Some of the combinations and permutations of vehicles, pigments and compositions which were used in the first part of this study are shown in the schematic below.

Exploratory Experimental Design



*Elvax, a copolymer of polyethylene and polyvinyl acetate.

Most of the coatings prepared for this first survey were found to be unsuitable for further study. The criteria of selection was based on good visual opacity and good mechanical properties. Several of the vehicle-pigment combinations were discarded because of the inability of the pigment to property crystallize and become opaque in the vehicle. Paraffin in polyethylene was an example of this type of behavior.

The polystyrene and the polyethylene-polyvinyl acetate (Elvax) vehicles were found to produce the best coating from considerations of their physical and optical properties when used with such pigments as eicosane (a twenty carbon normal hydrocarbon) or paraffin. Even in these two cases, there was considerable variation in the physical properties of the two coatings. The coating composed of 80% polystyrene-20% paraffin was hard and friable while the coating composed of 50% Elvax and 50% paraffin was tough and flexible. These coatings were studied further in order to determine the extent and direction of the change in their optical properties with temperature.

The coatings were prepared by heating the thermoplastic vehicle to just above its melting point and blending in molten pigment until a homogeneous solution of the two materials resulted. The molten mass was thencast into

the desired form. Usually sheets 10 to 20 mils thick were cast onto a highly polished silver plate for further tests.

Solar Absorptance Changes

The solar absorptance of these model coatings were determined by measuring the total hemispherical reflectance of the coating at several temperatures using a Cary 14 spectrophotometer with reflectance attachment. The response of a typical coating as a function of temperature is shown in figure 2. It is evident that this model coating undergoes a change, i.e. by a factor of two in its solar absorptance over the temperature range associated with the phase change of the paraffin in the polystyrene, namely, 65 to 70°C. The thermal emittance of this coating remained essentially constant at .93 throughout the whole temperature range studied.

The spectral reflectance of these coatings is shown in figures 3 and 4. The measurements were made at 25°C when the coatings were below their phase change temperatures. The absorption peaks in the near infrared region are due to the vibrations of the CH_2 and CH_3 groups in the crystalline paraffin. When the same spectral reflectance measurements were made at temperatures above the normal melting point of the paraffin, these peaks had diminished considerably and the whole reflection curve had shifted to higher values as the transparent coating allowed more light to reach the polished silver substrate and be reflected out of the coating.

The change in transparency of the two coatings with the same pigment, paraffin, was influenced by the nature of the vehicle. The coating which used polystyrene as the vehicle had a higher phase change temperature than one would have anticipated on the basis of the melting temperature of the uncompounded paraffin, namely, 55-60°C. This is an apparent anomaly for which there is no immediate explanation.

Thermal Properties

Measurements of the phase transition temperatures were made by using differential thermal analysis techniques and apparatus similar to those described by Vassallo and Harden (reference 2). A typical thermogram is shown in figure 5. The first peak at 57°C is characteristic of the transition at the melting point of paraffin vehicle. It is also close to the visual change in transparency for this coating. The second, broader peak at 120°C represents the melting temperature of the polyethylene-polyvinyl acetate copolymer (Elvax).

FUTURE WORK

This preliminary report has covered some of the early developments of our exploratory program to develop controllable thermal control coatings. In the near future we will be entering into the engineering development of the concept of using phase change coatings for thermal control. The engineering development will be carried out largely on a contracted basis and will include sutdies of:

(1) environmental stability

(2) variation of phase transition temperatures

(3) fabricating techniques

In addition, further fundamental studies will be undertaken in-house. More information is needed on the basic optical and thermal properties of these coating systems in order to understand how these coatings will stand up to long term operations with wide variations in heating and cooling rates. Microscopy studies will be initiated to provide a better insight into optimum size of dispersed pigments and their effect on the optical properties of the coating.

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1. G. W. Kuehl, U. S. Patent 2,710,274 (1955).

2. D. A. Vassallo and J. C. Harden, Analytical Chemistry, 34, 132 (1962).

CROSS-SECTION OF A PHASE CHANGE
ACTIVE THERMAL CONTROL COATING

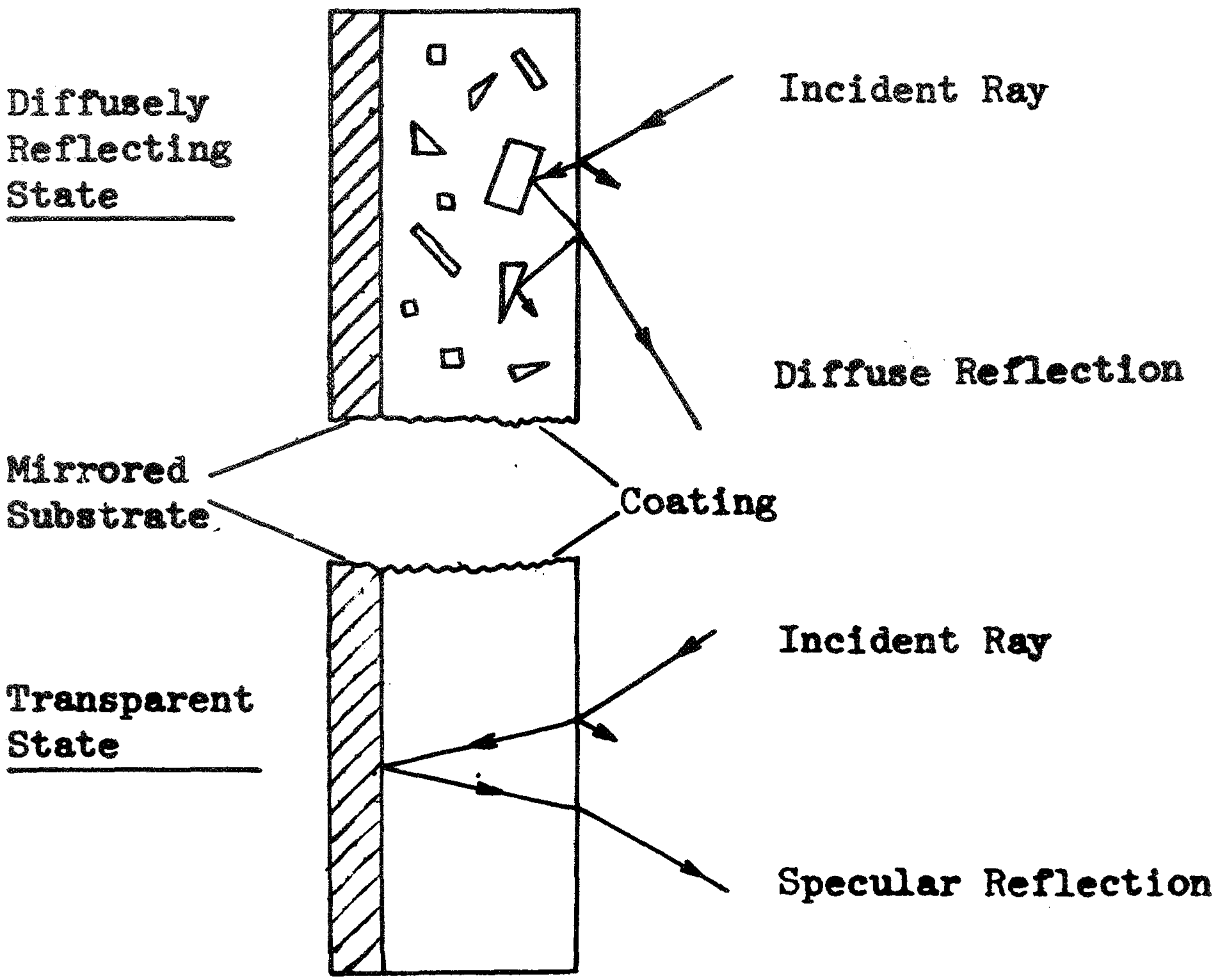

FIG. 1

CHANGE IN SOLAR ABSORPTANCE WITH TEMPERATURE
FOR PARAFFIN/POLYSTYRENE (20/80)

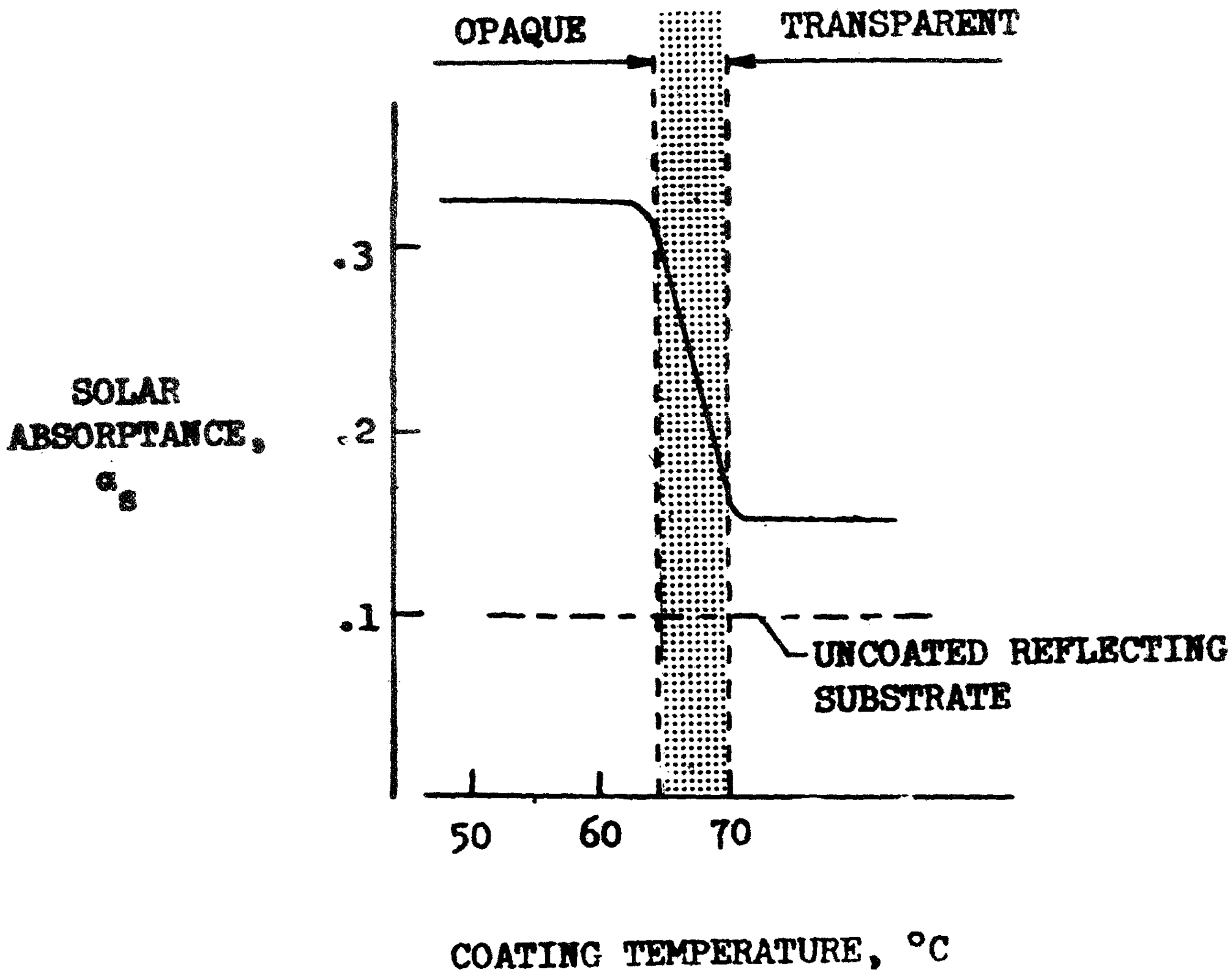

FIG. 2

REFLECTANCE CURVES FOR THE PARAFFIN-POLYSTYRENE COATING

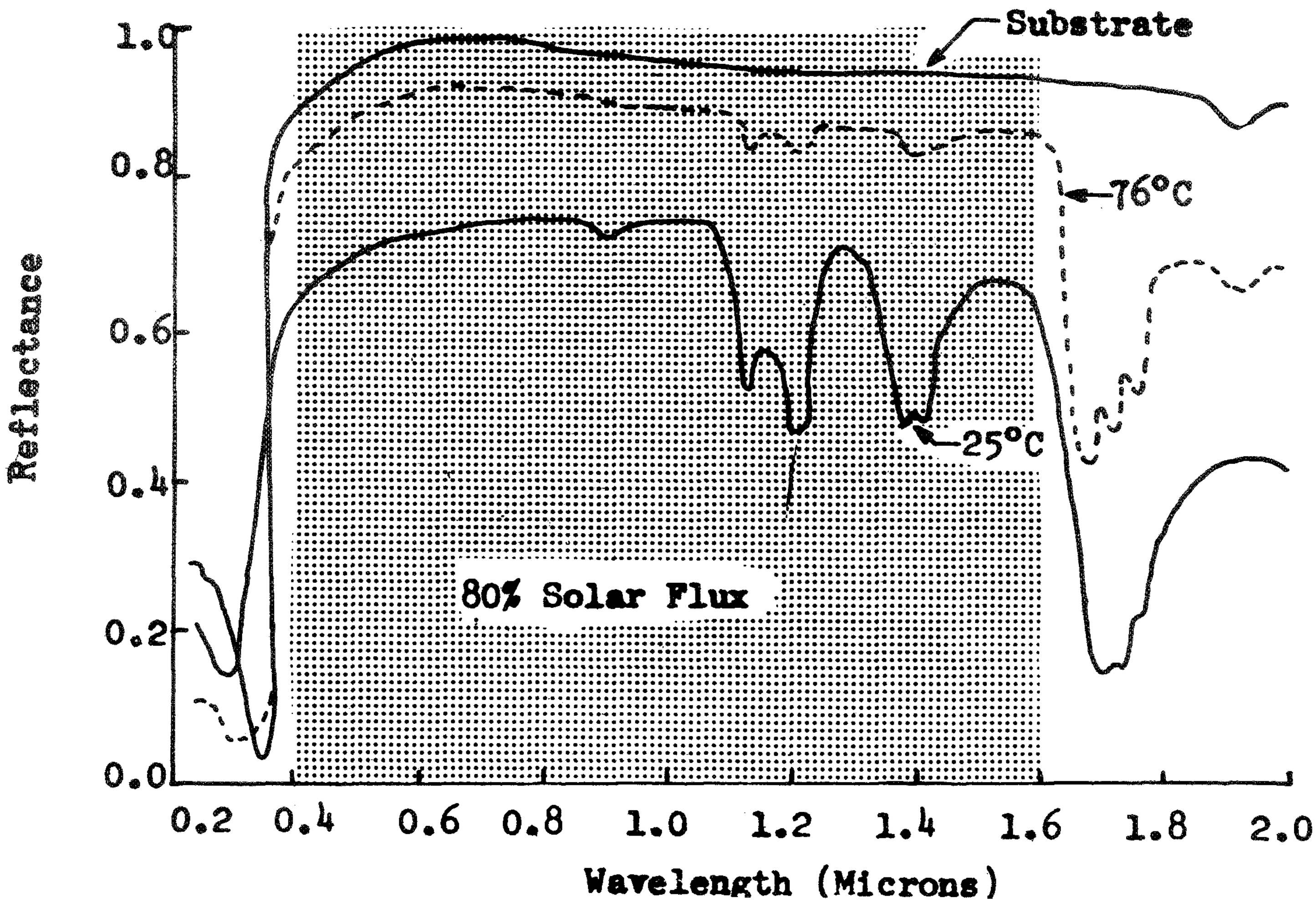

FIG. 3

REFLECTANCE CURVES FOR THE PARAFFIN-ELVAX COATING

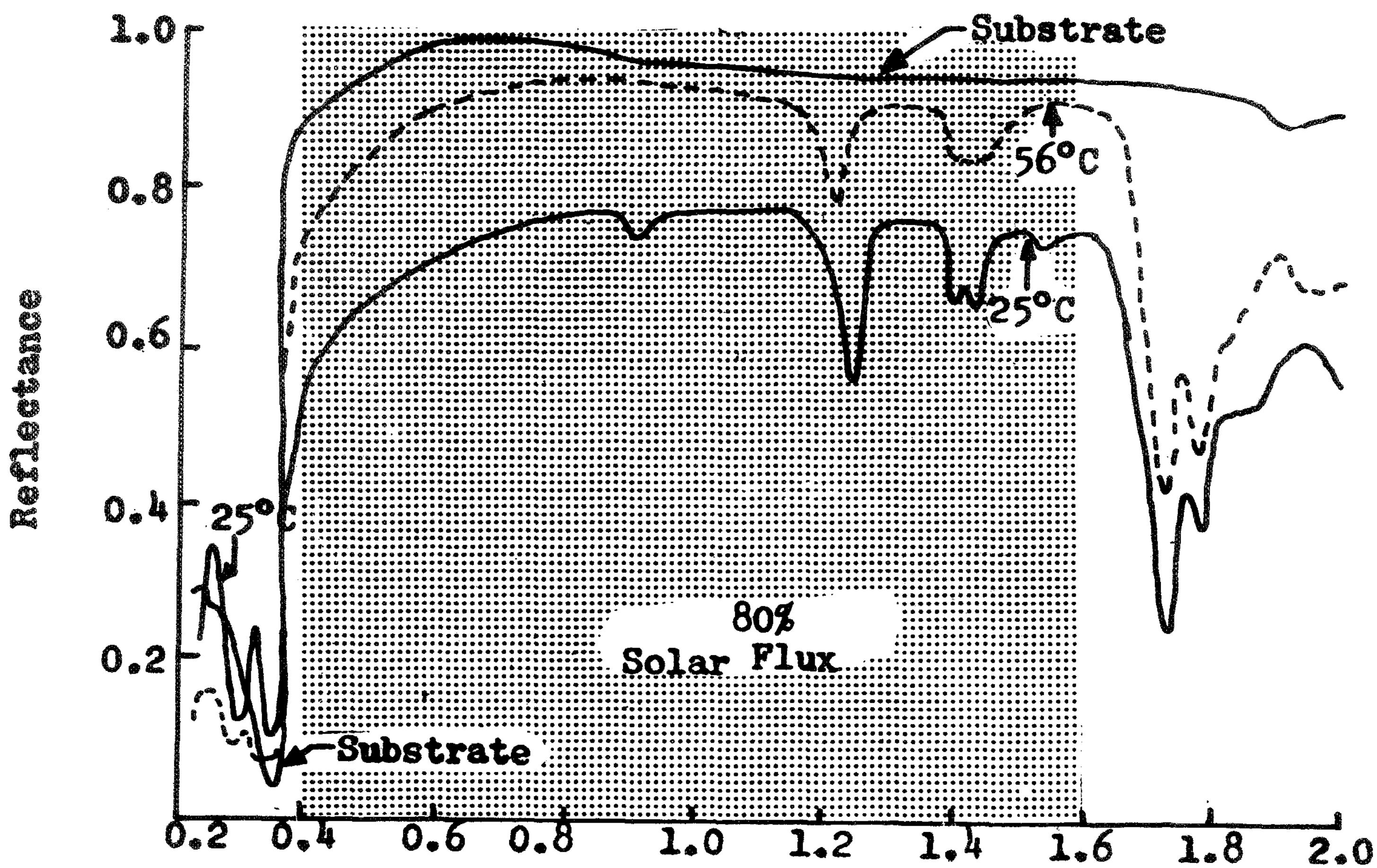

FIG. 4

DIFFERENTIAL THERMAL ANALYSIS
SCAN OF PARAFFIN-ELVAX COATING

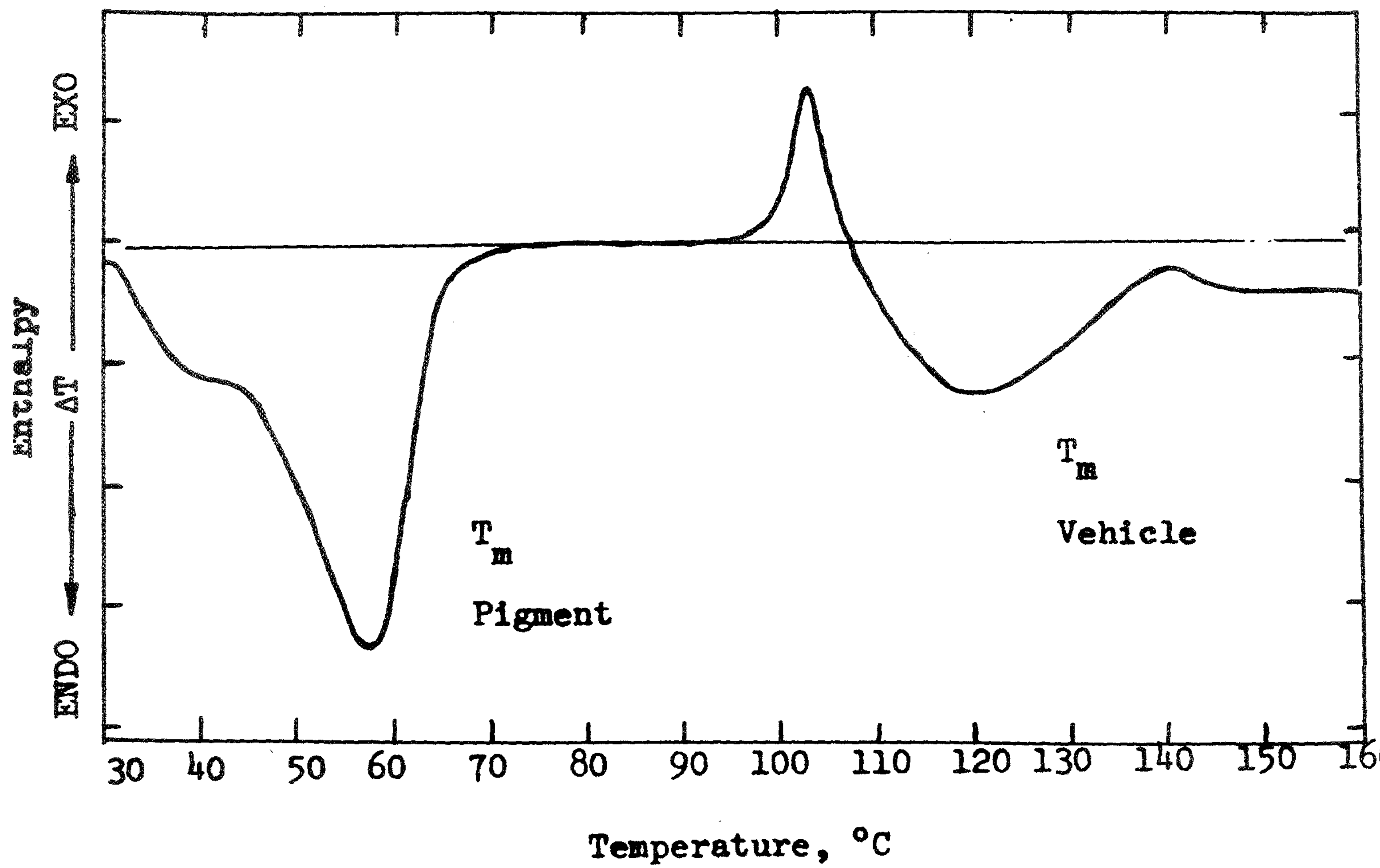

FIG 5

N66 32953

RESEARCH ON ACTIVE TEMPERATURE CONTROL AT GODDARD SPACE FLIGHT CENTER

by

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I. Design Philosophy of Goddard Space Flight Center In-House Satellites

The thermal design philosophy at Goddard Space Flight Center for in-house satellites has been to achieve temperature control primarily by passive means. This method has been successfully applied to spin-stabilized satellites operating under the following ground rules:

(1) For near earth satellites, where it is generally necessary to design for conditions of 60% to 100% sunlight, the structures have been restricted to nearly spherical configurations to minimize changes in absorbed sunlight with sun-spin axis angle.

(2) When highly non-spherical structures have been required, the orbits have been highly elliptical, so that the design could be based on absorption of continuous sunlight varying in magnitude only with changes in the attitude of the spin axis relative to the sun.

(3) Internal power dissipation has either been negligible compared to the total absorbed and radiated power or it has been possible to design on the basis that the power is dissipated at a constant rate.

(4) For design purposes, shadow periods have been assumed to be limited to approximately one hour or less. It has not been necessary to maintain temperature limits for longer shadow periods although 2-1/2 and 9-hour shadows have occurred on Explorers 14 and 18, respectively.

(5) It has been possible for highly elliptical orbits to control the launch time in order to restrict the range of possible sun-spin axis angles or delay for many months the occurrence of long term shadows.

(6) Most of the electronic components and experiments have been designed and tested to withstand a relatively broad temperature range.

II. Need for Active Temperature Control on Future Goddard Satellites

If restrictions on configuration, orbit, launch window, and shadow periods are lifted, it is not possible to control temperatures passively within the limits of approximately $-10^{\circ}C$ and $+50^{\circ}C$. With increasing emphasis being placed on reliability, some device such as an active controller must be employed to keep temperature excursions to a minimum.

III. Characteristics of Existing Active Control Systems

A. Louvers - Where louvers are used for temperature control, they are generally located on surfaces which are shaded from the sun. However, it is generally impossible to find surfaces which are always shaded on spin stabilized spacecraft. With solar influx on louvered skins, wide variations of effective solar absorptance with shutter and solar aspect angles can occur (see Figure 1). This is largely due to the cavity effect at small louver angles. This problem makes it difficult to design a system which will emit nearly linearly (see Figure 2) with shutter position, while subjected to non-linear solar heating. Most of these systems also require a high degree of conductive coupling between the electronic components and the radiating surfaces.

In addition, it is not entirely certain that louvers which are oriented parallel to the spin axis and which are located at the spacecraft periphery will not affect the spin rate when they open or close.

B. Internal Canister - The Telstar satellite employed a cylindrical canister in which the electronic components were housed. The sides were wrapped with super insulation and the ends were used as active control surfaces. By varying the positions of the end caps, the radiation exchange with the spacecraft skin could be

controlled. The ratio of the volume of the internal canister to the volume of the total spacecraft was relatively small so that the temperature of the internal canister was almost independent of temperature gradients along the skin of the spacecraft.

C. Internal Shutter - The Relay satellite employed an internally mounted rotary shutter which controlled the amount of heat radiated from the electronic components to the bottom surface of the spacecraft. The bottom surface was kept relatively cool by controlling the launchtime so that the spacecraft's spin axis remained nearly perpendicular with respect to the sun. The temperature sensor for controlling the shutter was located at the battery and thus responded to local rather than average spacecraft temperature changes.

D. Rotary Blades Mounted on Skin (Atlas Able) - Spacecraft internal temperatures can also be controlled through the use of rotary blades which alternately expose or cover different surface coatings. This system was first designed for an Altas-Able satellite but was never successfully flown because of launch vehicle failures. It employs a light weight blade, bearings, and a bimetallic sensor-actuator. It is sensitive to inertial and vibratory loads and its calibration is easily shifted through handling and repeated cycling. Of all of the systems which have been discussed, the rotary blade controller appears to be the most suitable for controlling temperatures of spin-stabilized, non-oriented, satellites.

IV. Why Goddard's Requirements for Active Control are Different from Those of Other Spacecraft

By and large, the Goddard in-house satellites are spin stabilized with solar influx impinging on all surfaces of the skin at some time during the spacecraft's lifetime. This precludes the use of any one surface that is always shaded for mounting a louvered or internal shutter system. The high packaging density of Goddard spacecraft prohibits the use of the internal radiative technique for an active control system. In addition, packaging methods presently employed at GSFC make it difficult to control heat flow by conduction. Electronic modules are sometimes stacked six to eight inches above heat dissipating surfaces, with poor conductive coupling between layers.

V. Work to Date on Active Controller at GSFC

Because of the previously mentioned limitations of louver, canister, and shutter designs when applied to Goddard satellite configurations, it was decided to investigate the characteristics of an improved version of the Rotary Blade Controller. A new system is being designed employing a bimetallic coil with a high spring constant and molydisulfide impregnated nylon thrust bearings inserted to give the assembly more axial stability under vibratory loads. Figure 3 shows the controller in its mounting fixture.

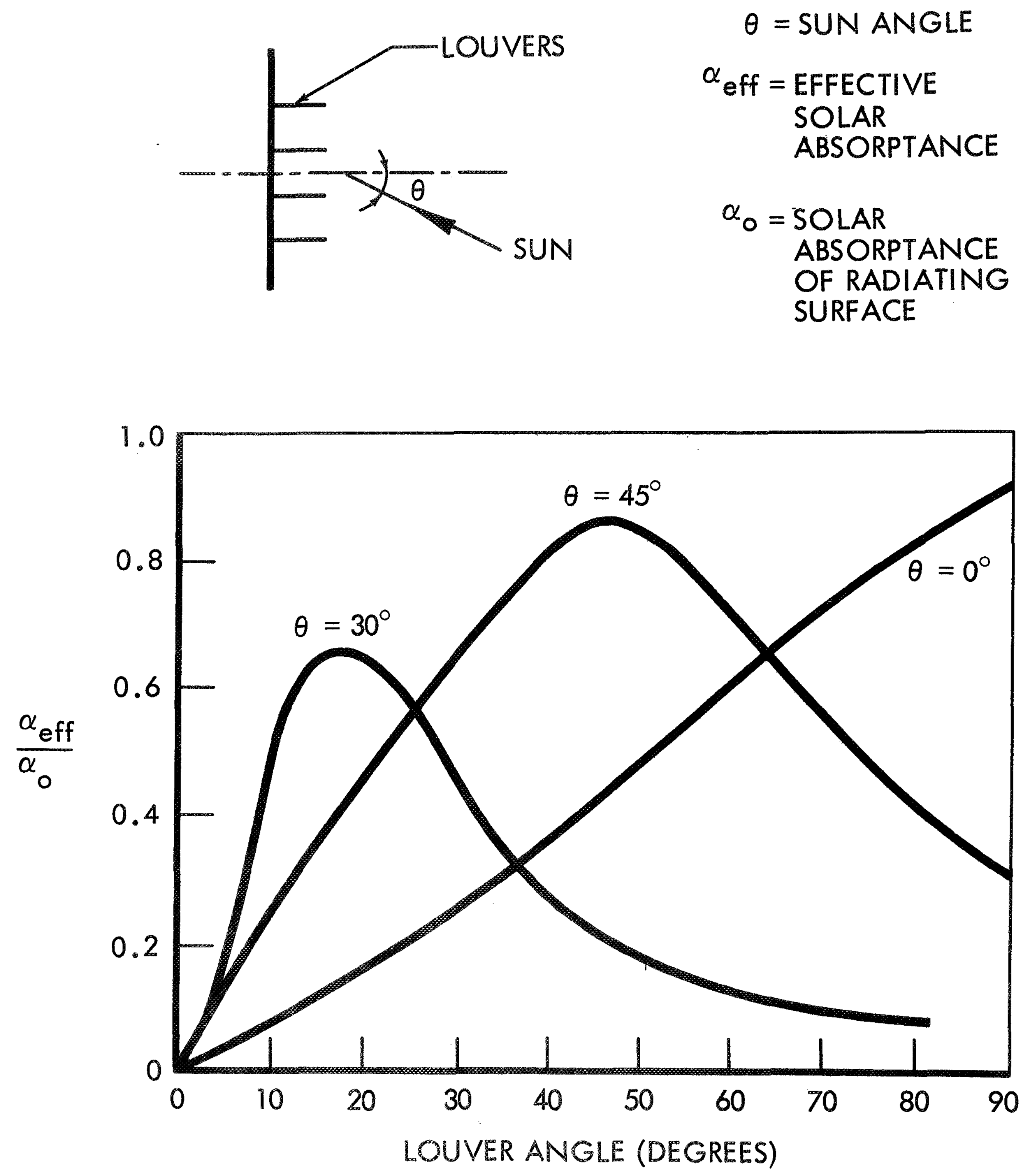

Figure 1. Effective Absorptance Vs. Louver Angle For Various Sun Agnles (θ)

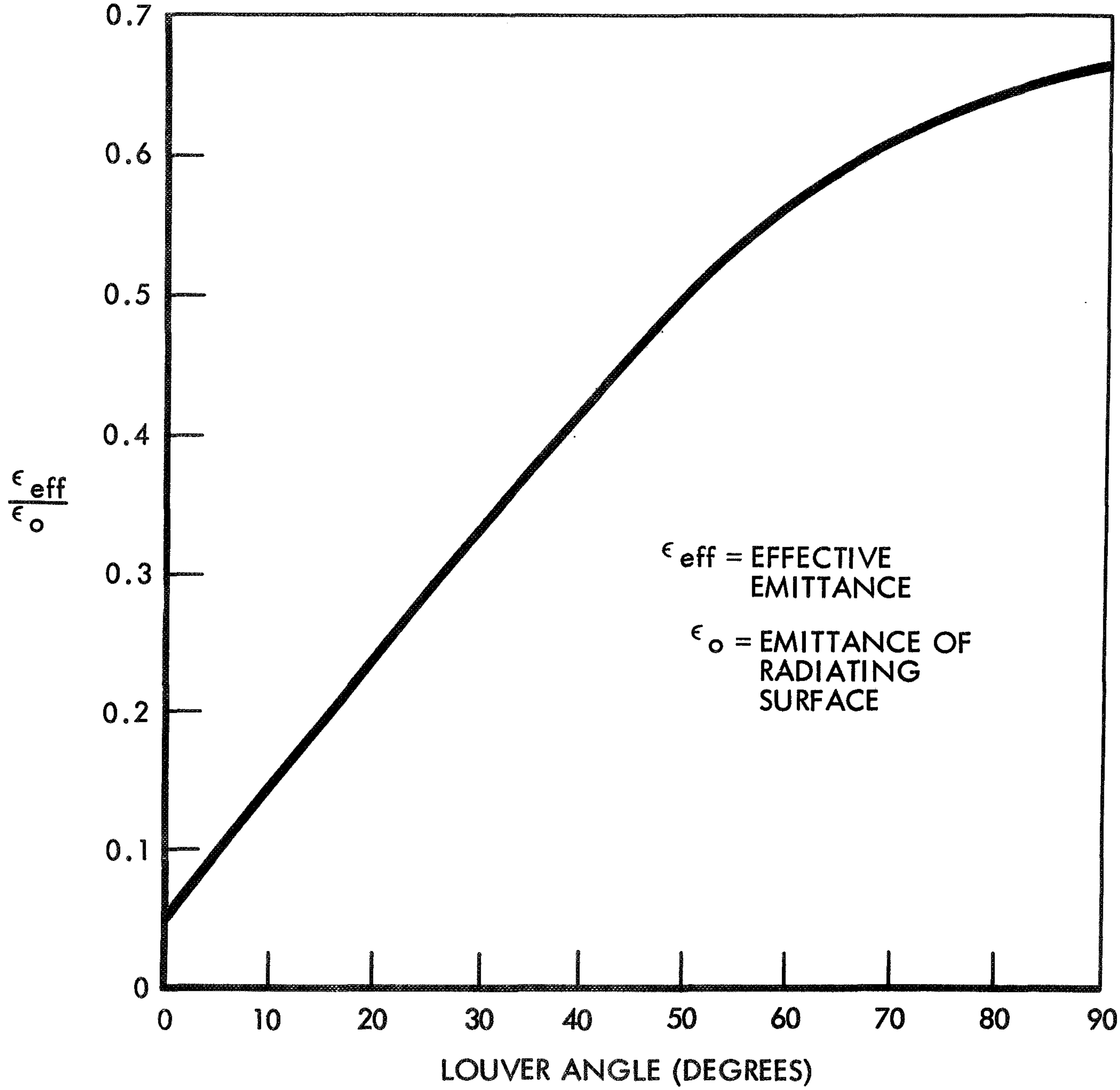

Figure 2. Effective Emittance Vs. Louver Angle

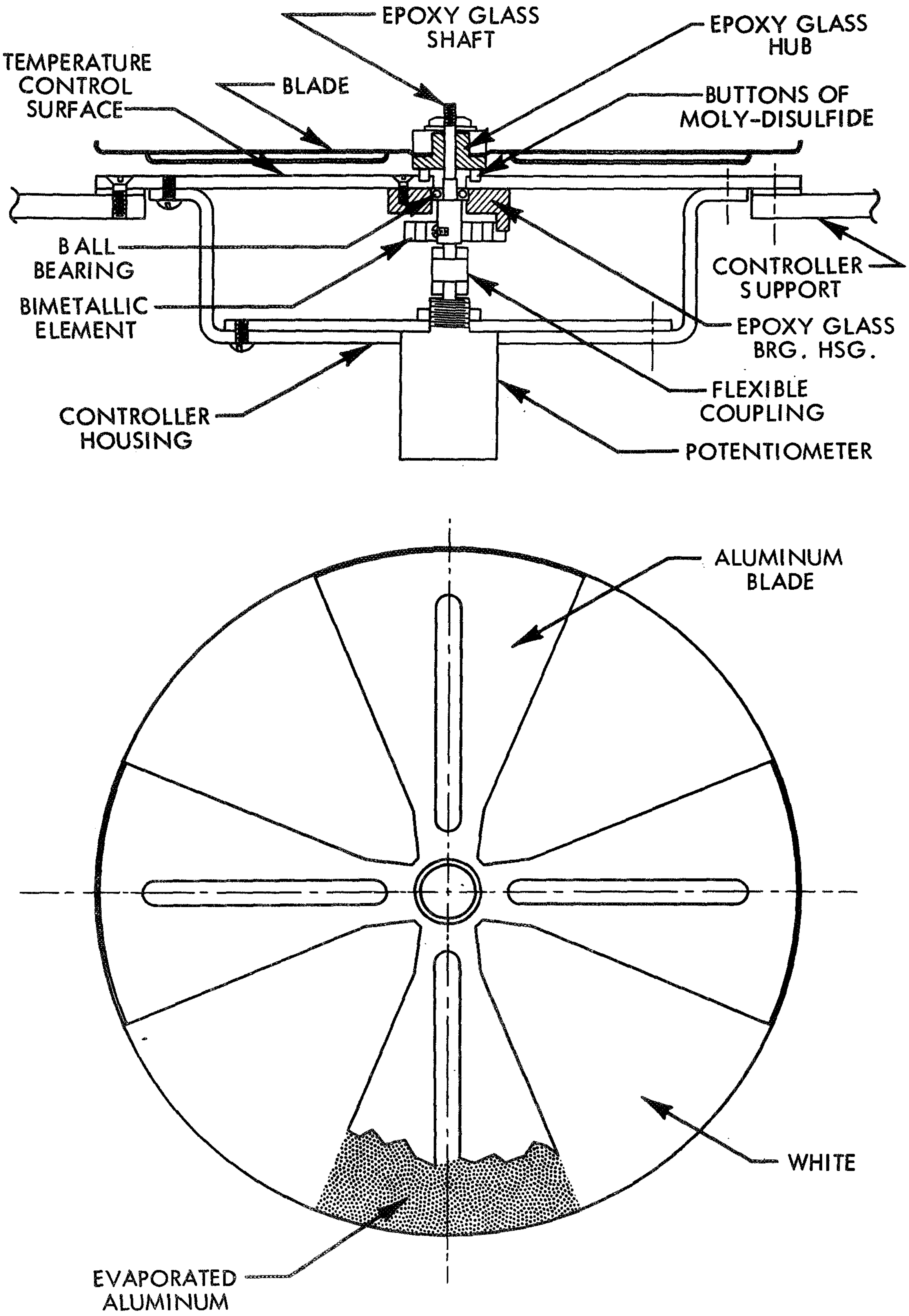

Figure 3. Active Temperature Control Assembly

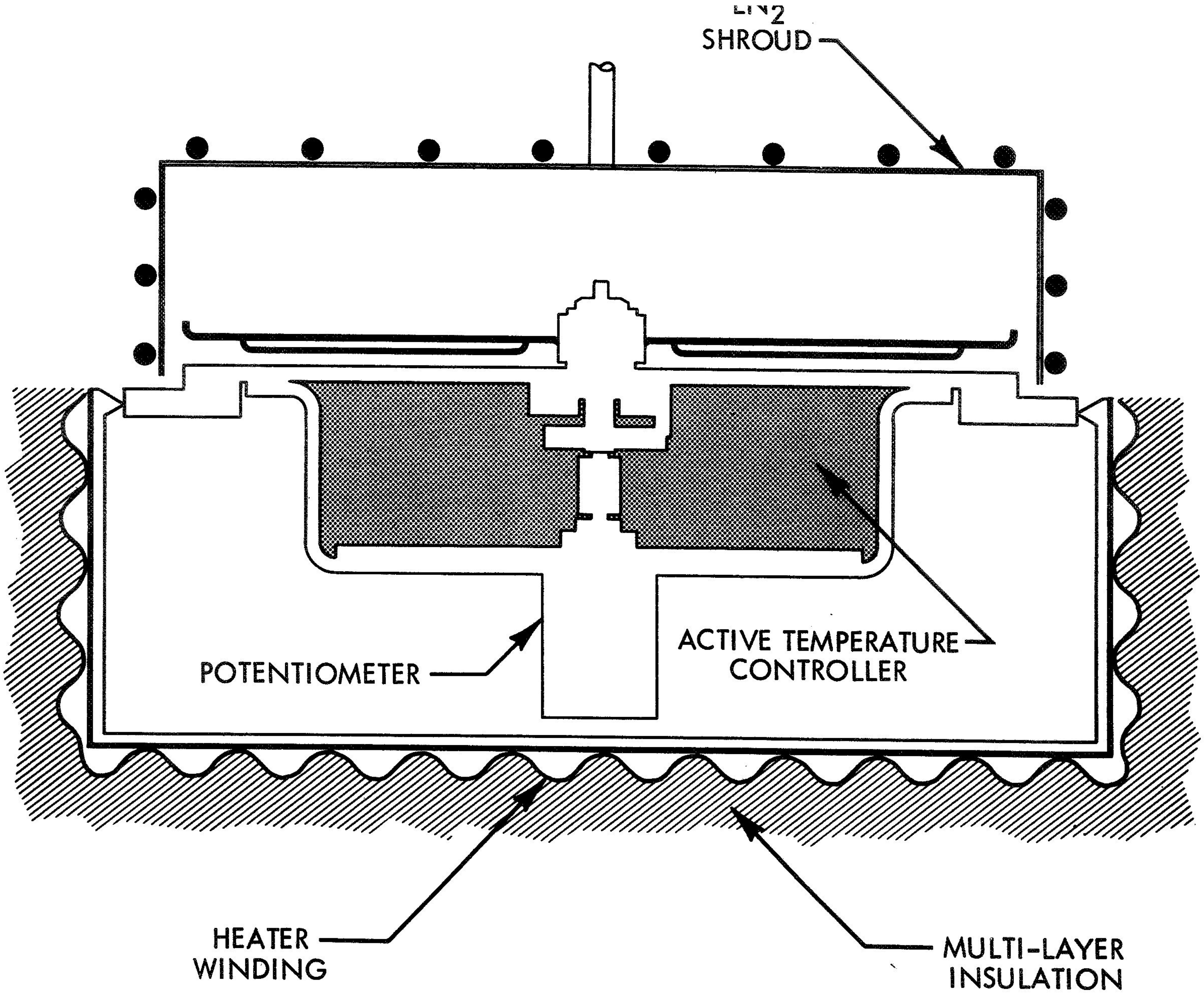

Figure 4. Assembly of Temperature Controller Into Test Fixture

N66 32954

TEMPERATURE CONTROL LOUVERS FOR THE MARINER VENUS AND MARINER MARS SPACECRAFTS

M. Gram - JPL

The function of active temperature control devices is to suppress temperature excursions and provide tighter temperature regulation than would otherwise result. Were it not for the widely varied heat inputs to spacecraft subassemblies, and for the bearing which temperatures have upon reliability and endurance, active temperature control devices would not be required.

Louvers are but one type of active temperature control device. Figures I and II show the louver systems designed for Mariner R and C respectively. Whereas they differ somewhat in configuration, mechanically and functionally they are similar. Mechanical features common to both systems are listed in Table I. An overall comparison is provided in Table II.

Thermal performance for the two systems in terms of the effective emittance as a function of louver blade angle is given in Table III. Theoretical performance values are also given for comparison (ref. JPL TR 32-555, Analysis of Movable Louvers for Temperature Control, J. Plamondon, 1964). The theoretical values have been based on diffuse emission-reflection, infinite length louver blades, and assumes no heat loss from bracketry.

In order to rationalize empirical and theoretical performance figures, an adjustment has been made to the empirical values, forcing them in agreement with theoretical values for the fully closed louver condition. The adjustment (or tare) may be considered to be an area of unit emittance which radiated in parallel with the louvers. The adjusted thermal performance figures are given in Table IV, and as can be seen are in only fair agreement with the theoretical values. The differences are felt to stem primarily from experimental errors made during the measurement of louver performance, and to a lesser extent from the inexact nature of the mathematical model.

TABLE I

Mechanical Features of the Mariner Louver System

1. Louvers are individually actuated--not ganged.
2. Louver sensing and actuating elements are spiral-shaped bimetal coils.
3. The bimetal sensor-actuator is primarily radiatively coupled with the "sensed" temperature.
4. Louver blades are made of thin gauge polished aluminum alloy sheets.
5. Louver blades are center pivoted (1) to better withstand dynamic environments, and (2) to permit their usage in any spacecraft attitude during test.
6. Louvers are supported in bushing type bearings.
7. The temperature for incipient opening of the louvers may be varied by adjusting the anchor point of the bimetal coil.
8. Individual louvers may be removed from the assembly easily for replacement, cleaning, or inspection without affecting the louver adjustment.

TABLE II

Comparison of Mariner II and Mariner C
Louver Assemblies

		MA-II	MA-C
1.	Effective emittance closed	.08	.12
2.	Effective emittance open	.72	.76
3.	Area of louver assembly	1.76 lb/sq. ft.	1.62 sq. ft.
4.	Weight	2.20 lb.	1.35 lb.
5.	Weight per area ratio	1.76 lb/sq. ft.	.83 lb/sq. ft.
6.	Actuation range	$30^{\circ}F$	$27^{\circ}F$
7.	Year designed	1961	1963
8.	Year flown	1962	1964 (?)
9.	Static bearing friction in one-G field	9° angle	6° angle
10.	Louver thickness	20 mil	2 layers 5 mil
11.	Attachment to chassis	rivet	bolt

TABLE III

(LOUVER PERFORMANCE)

Louver Angle	Effective Emittance		
	Mariner II	Mariner C	Theoretical
0°	.08	.12	.03
30	.37	.57	.37
60	.61	.71	.57
90	.72	.76	.63

TABLE IV

(ADJUSTED LOUVER PERFORMANCE)

Louver Angle	Effective Emittance		
	Mariner II	Mariner C	Theoretical
0°	.03	.03	.03
30	.32	.48	.37
60	.56	.62	.57
90	.67	.67	.63
Tare	9.0 Sq. In.	21.0	* --

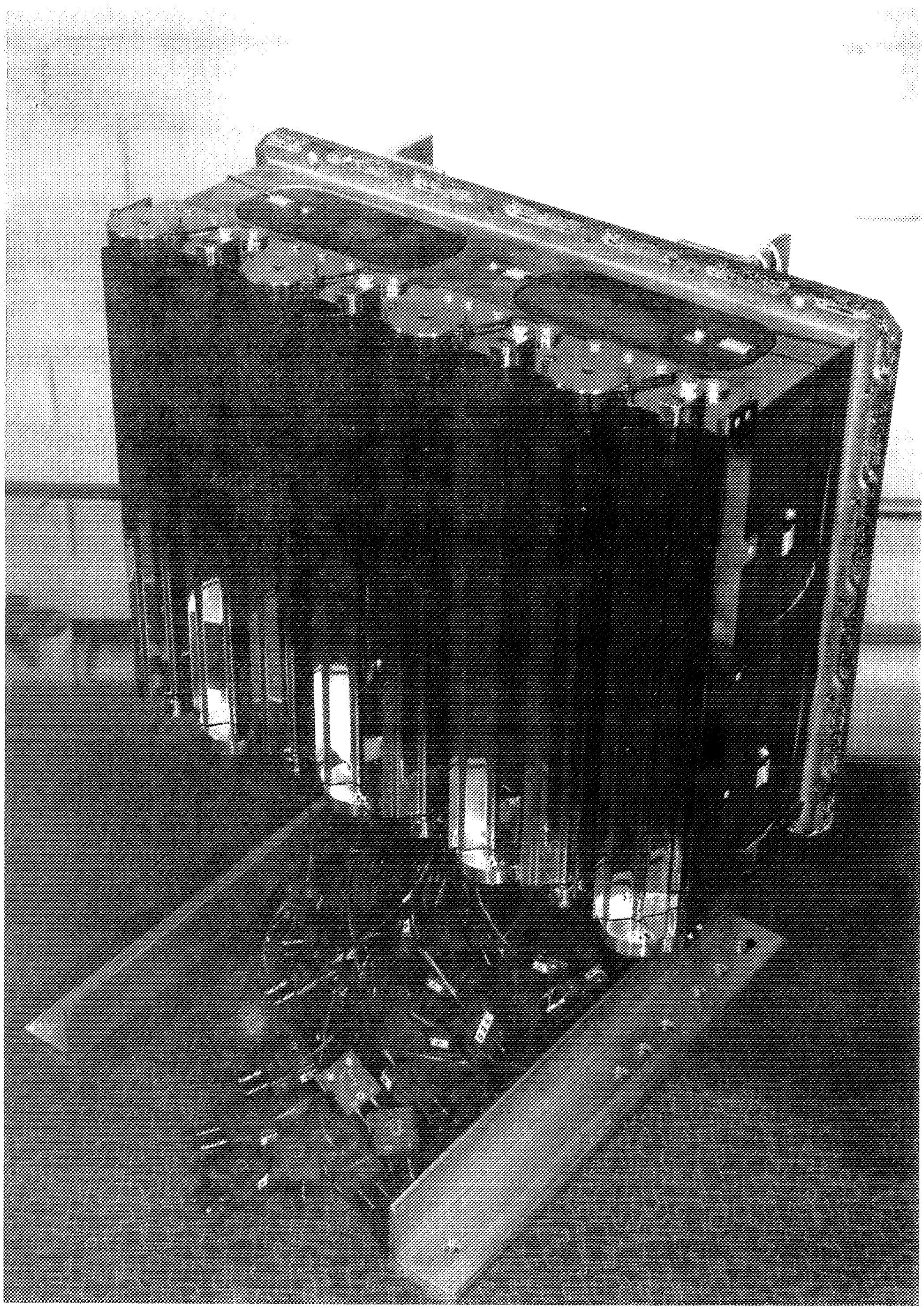

Fig. 1

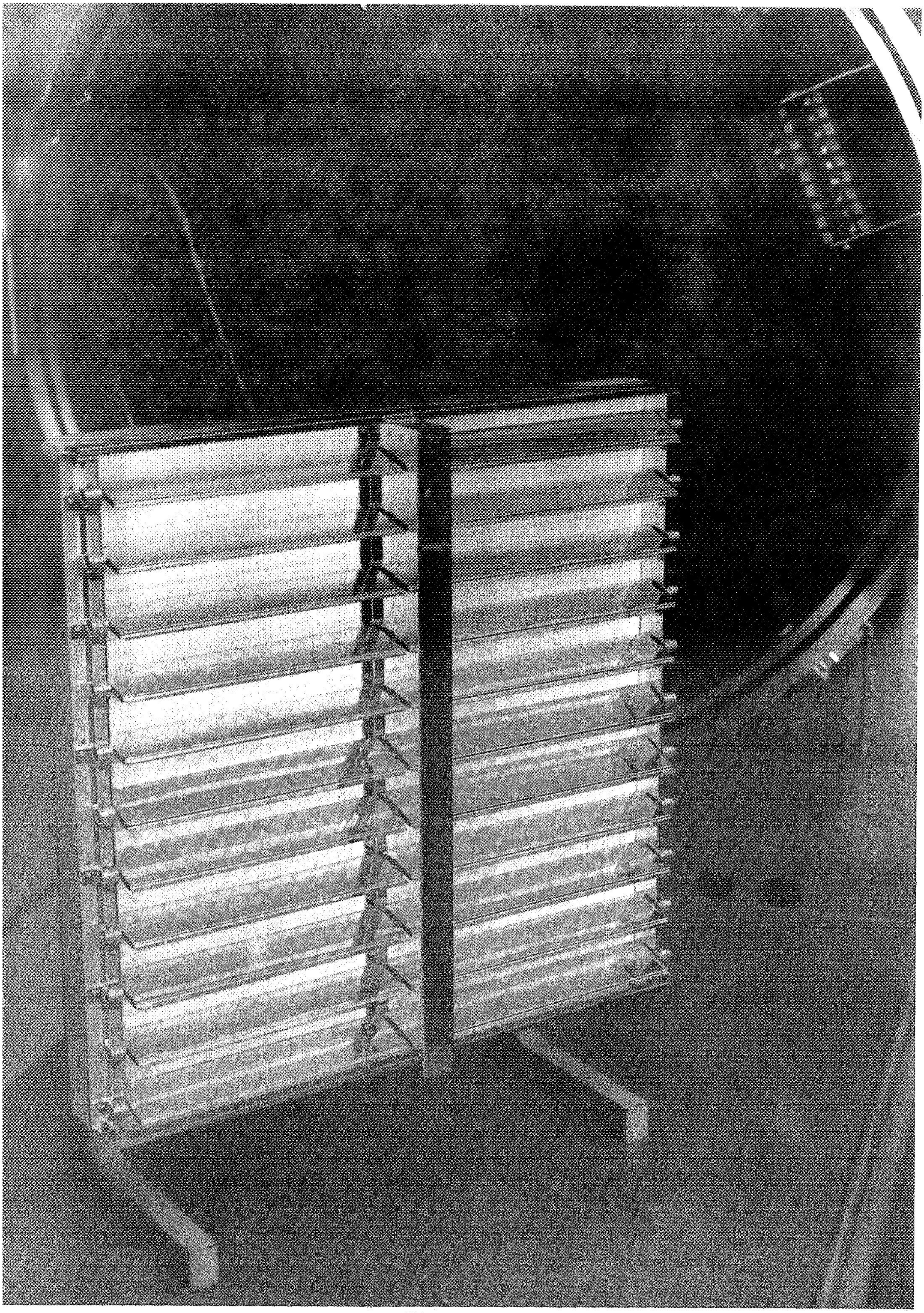

Fig. 2